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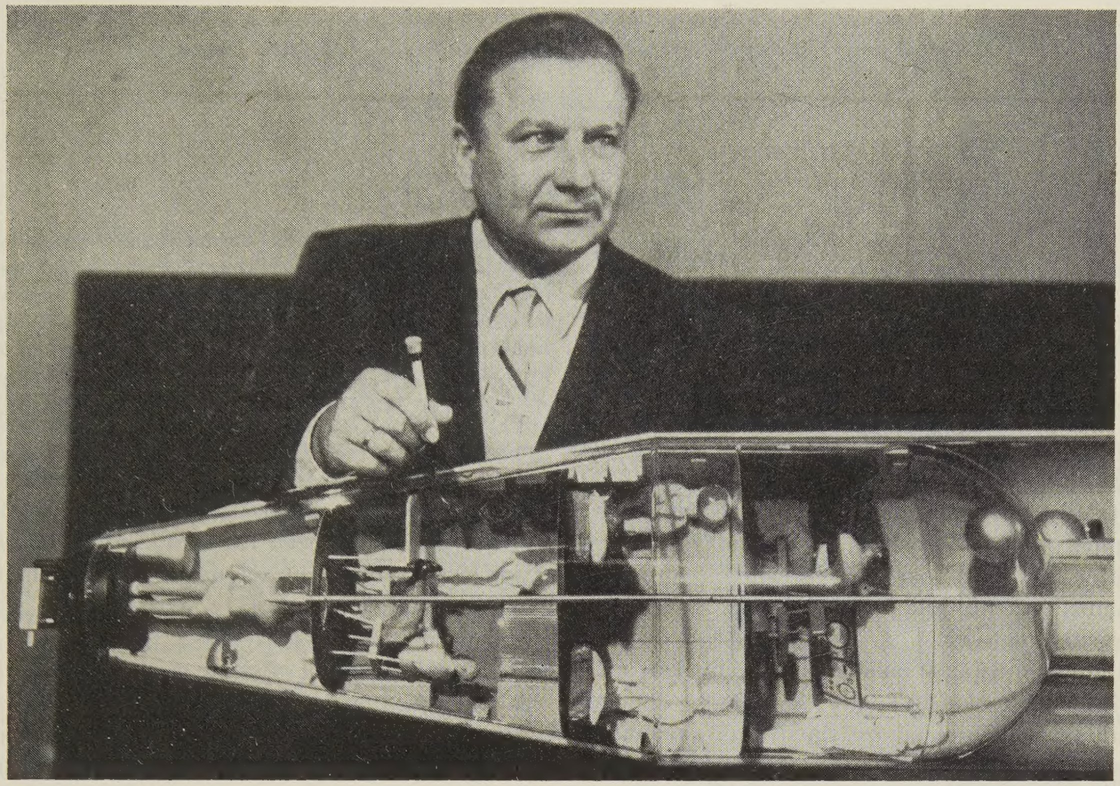

Krafft A. Ehricke

Krafft A. Ehricke was born on March 24, 1917, in Berlin, Ger. In April, 1942, he received the equivalent of the M.S. degree in aeronautical engineering from the Technical University of Berlin. While earning this degree he also studied celestial mechanics and nuclear physics at the University.

He became a research engineer for German Army Ordnance in June, 1942, and worked on the V-2 rocket program at Peenemuende until May, 1945. Following World War II, along with other German scientists and engineers, he was invited to participate in the U. S. Army's missile and rocket development program. From January, 1947, to September, 1950, he worked on advanced V-2 systems analysis and ramjet projects at Fort Bliss, Tex. In September, 1950, he was transferred to the Army Ballistic Missile Center at Redstone Arsenal, Huntsville, Ala., where he served as chief of the gas dynamics section until November, 1952. At that time he joined the preliminary design department of Bell Aircraft Corp., Buffalo, N. Y., where he became assistant project engineer on a long-range glide rocket study. In November, 1954, he joined Convair, starting as a preliminary design specialist on the San Diego division's Atlas intercontinental ballistic missile project. This project later resulted in formation of Convair's Astronautics division. He became chief of preliminary design and systems analysis for Convair-Astronautics in January, 1956; in June, 1958, was named assistant to the chief engineer; and became Program Director of Project Centaur in January, 1959.

He is a consultant to the Office of the Secretary of Defense. He holds memberships in the Institute of Aeronautical Sciences, American Rocket Society, and German Society for Space Research. He is a Fellow of the British Interplanetary Society. He received the 1957 Astronautics Award of the American Rocket Society for "outstanding contribution to the advancement of space flight." In 1955 he won the Guenther Loeser medal for presentation of the best paper at the sixth Congress of the International Astronautical Federation in Copenhagen, Denmark. Presently he is working on a three-volume textbook on space flight for the D. Van Nostrand Co., Inc., New York, N. Y. He teaches space flight mechanics and space navigation at the San Diego State College and conducts private research on space flight.

Mr. Ehricke became an American citizen in 1954.

Guest Editorial

KRAFFT A. EHRICKE

THE PGMIL TRANSACTIONS are on record of keeping their readers informed about the vital problems which confront us at the dawn of astronautics. Scientific, technical, economic and military ingredients are involved in practically every problem area, often in such a complex fashion that it is difficult to decide whether the particular effort toward a solution is a scientific, a military, or—easiest of them all—a technical endeavor. This is so because space flight is a living concept, of concern to all, not just to individual professions. Today, the living concepts are almost all induced by the titanic struggle of ideas and philosophies among men. More than war, they are the most fruitful kind of conflict, as they challenge the moral and the idealistic strength of the nations involved. War forever remains a fundamental threat, and adequate defense must be maintained at all cost. However, it is no longer enough to hold one's own. The perhaps bitter, but immensely progressive law that not growing (somehow) means dying, is back in the 20th century with a ruthlessness and finality that reminds one of the Roman times.

There are many important ways in which this nation must grow today, if the civilization for which it stands is to survive. Among the most important ones is our growth into a spacegoing nation. The vast environment of cislunar and interplanetary space, with the still largely unknown worlds and forms of life in it, are rapidly becoming accessible to us. They wait to be explored, as the new world of the renaissance waited for its discoverers and pioneers. It is quite apparent to everyone today that we must not deny ourselves the potential of this knowledge.

The fundamental basis of space flight is energy and its conversion into a form which drives spacecraft. In the course of missile development, the most rudimentary type of rocket propulsion, namely, the chemical rocket engine, is being developed to comparative perfection. However, chemical energy is inadequate for space technology in the long run. We make our first excursions into space with chemical propulsion systems, as we used balloons at the dawn of aeronautics. For a fully developed aeronautical technology, capable of serving man's need for rapid global transportation, the combustion engine is a prerequisite. For a similar development in astronautics we need nuclear drives.

The multitude of possible nuclear drives is briefly reviewed in the first introductory article. Among them, the electrostatic and the plasma drives are most promising for the attainment of specific impulses in excess of 2000 lb (thrust)/lb/sec (expelled material). Their low thrust power yields very long interplanetary mission periods. Therefore, initial applications are likely to pertain to operations in terrestrial and cislunar space, for example, to constellational stabilization of multi-satellite systems, and to the interorbital transport of heavy payloads. For such missions the time factor is not critical, and much lower thrust-to-weight ratios are acceptable than for manned interplanetary flights. For example $3 \cdot 10^{-7}$ g (1 foot per second per day) is acceptable for satellite stabilization, while for interplanetary missions one must strive at least toward 10^{-4} g. Given proper emphasis, these "first generation" electric drives might be used for satellite stabilization (in 8- to 24-hour orbits) beginning in 1963–1965 with thrust values of the order of $3 \cdot 10^{-3} - 10^{-1}$ lb. In the 1965–1970 period, thrust values of 0.5 to 30 lb can be expected to become practical for ion drives and possibly higher values for plasma propulsion systems.

Electrical and electronic technology will be mated with energy conversion and propulsion technology to open the entire solar system, and perhaps the neighboring part of our galaxy, to man. To this important subject the present issue is dedicated.

Some Problems in Ionic Propulsion Systems*

E. STUHLINGER† AND R. SEITZ†

Summary—Some of the problems and applications of ionic propulsion systems are discussed. Three different systems' optimization criteria are considered: the maximization of the initial acceleration of a space vehicle; the minimization of the total-mass-to-payload-mass ratio; and the minimization of the propellant mass required to refuel the vehicle. The production, acceleration, and neutralization of beams of singly-ionized cesium ions is also discussed in limited detail. A hot tungsten contact-catalyst type of ion source is assumed and some experimental results with such a source are reviewed. Finally, a simplified treatment of the space charge neutralization of a positive ion beam in the region behind the space vehicle is presented. In this treatment, the positive-ion beam is replaced by an infinitely-long cylinder of uniformly-distributed positive charge. Electrons are emitted from an annular filament encircling the perimeter of the beam. It is shown that this approximation leads to radial oscillations of the electrons through the positive column.

List of Symbols

a = radius of positive ion beam, meters
a_i = initial acceleration of the space vehicle, g's
e = absolute magnitude of electronic charge, coulombs
I = ionization potential, volts
j = current density, amps/meters
k = Boltzmann Constant, joules/°K
L = electrical power output of vehicle powerplant, watts
M_P = propellant mass, metric tons
M_L = payload mass, metric tons
M_0 = total initial mass of vehicle, metric tons
m = particle mass, kg
m_e = mass of the electron, kg
m_i = mass of the ion, kg
$\dot{m}$ = rate of ejection of propellant mass, kg/sec
n_+/n = the ratio of cesium ions to the neutral atoms evaporating from a hot catalytic surface with which they are in thermal equilibrium
p_+/p = the ratio of the statistical weights of cesium ions in the ground state to neutral atoms in the ground state as they evaporate from the hot surface
P = thrust force per unit area for an ionic thrust unit, newtons/meters^2
$R = r_0/a$
r = radial distance from the axis of the positive ion beam, meters
r_0 = radial distance of the electron emitters from the axis of the ion beam, meters
T = temperature, °K
T_c = critical temperature (catalyst temperature at which the ionization efficiency has dropped to 50 per cent), °K
V = Coulomb potential, volts
v_{ex} = exhaust velocity of ion beam, meters/sec (unless otherwise specified)
W = kinetic energy of electrons, joules
X = distance to a point on the axis of the diverging ion beam, measured from the point at which the ion beam has its minimum diameter, meters
x = interelectrode spacing, meters
α = power-to-mass ratio of the electrical powerplant, kw/kg
β = mass-to-thrust ratio of the ionic thrust unit, kg/"kg" (1 "kg" = 9.8 newtons)
ϵ_0 = permittivity of free space, mks units
η_i = ratio of the vehicle initial acceleration to its maximum possible value
λ_+ = linear charge density of the positive ion beam, coulombs/meter
λ_+' = linear charge density of the positive ion beam corrected for partial neutralization by electrons, coulombs/meter
ϕ = work function of catalyst, volts
ρ = volumetric charge density, coulombs/meter^3
τ = total operating period of the ionic propulsion system, seconds.

I. Introduction

As a result of the intrinsically-high exhaust velocities which ionic propulsion systems can provide, a certain amount of interest has developed in recent years in the possibilities of their application for low-thrust space flight. The importance of high exhaust velocities for rocket-type propulsion systems may be seen from the following consideration. The thrust force produced by a rocket motor is given by $\dot{m}v_{ex}$, where $\dot{m}$ is the rate of flow of the propellant mass, and v_{ex} is the propellant ejection velocity relative to the rocket. If a constant thrust force is to be maintained, an increase in exhaust velocity will result in a decrease in the mass ejection rate of the propellant, improving propellant consumption economy.

This seems to imply that the exhaust velocity v_{ex} should be as high as possible for any given set of space vehicle parameters. However, there exists an optimum exhaust velocity at which the total mass of the vehicle is a minimum. While the thrust of the rocket motor increases with v_{ex}, the power which must be supplied to the rocket exhaust increases with v_{ex}^2. Therefore, the mass

* Manuscript received by the PGMIL, March 5, 1959.
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of the powerplant, which is roughly proportional to its power output, increases more rapidly than the thrust and soon becomes the governing factor in determining the total mass of the vehicle. Consequently, a compromise must be worked out between the propellant economy which demands large values of v_{ex}, and the power economy which demands lower values of v_{ex}. Typical optimum exhaust velocities will be of the order of 50 to 120 km per second.

II. Basic Components

A variety of different ionic propulsion systems is possible. Generally, however, they all contain a thermal power source, a thermal-to-electric-power conversion system, a radiator which serves as a low temperature heat dissipator, and the ionic propulsion unit itself.

One promising system would utilize a nuclear reactor-turbogenerator combination. A number of authors have dealt with propulsion systems of this type [1]–[9]. A complete ionically-propelled space ship capable of traveling to Mars and back, together with design and performance data, has been proposed by Stuhlinger [8], [9]. A typical set of parameters for such a space ship is given in Table I and a schematic drawing of the propulsion system is shown in Fig. 1. Typical reactor-turbogenerator power supplies for space applications have been widely discussed and will not be treated in this paper.

TABLE I

Design and Performance Data of a Representative Space Vehicle for a Return Trip to Mars

Basic Assumptions	
Total travel time	3 years
Payload	250 tons
Ionic mass (cesium)	2.2×10^{-22} g
Initial acceleration	5×10^{-5} g
Specific power of powerplant	0.1 kw/kg
Specific mass of thrust unit	3.64×10^{3} tons volt$^{3/2}$ amp^{-1}
Calculated Design Data	
Total initial mass	600 tons
Propellant mass	150 tons
Thrust unit mass	20 tons
Powerplant mass	180 tons
Total electric power	20 megawatts
Power contained in jet	18 megawatts
Driving voltage	9900 volts
Total ion current	1820 amperes
Exhaust velocity	120 km sec^{-1}
Characteristic velocity	67 km sec^{-1}
Thrust	30.6 kg

Weight-to-power ratios, including the weight of the radiator and the shield, of from 10 to 20 lb/kw have been quoted for auxiliary power supplies in the megawatt range. Further research and development work will be necessary before these power supplies are sufficiently reliable to operate for a year or more with little or no maintenance at the weight-to-power ratios quoted above.

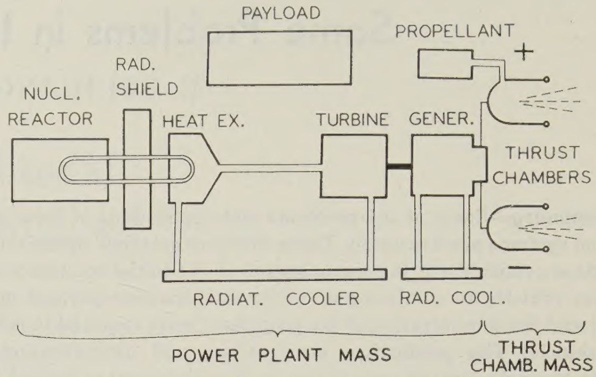

Fig. 1—Block diagram of an ionically-propelled space ship.

A radiator must be provided to serve as a low-temperature heat dissipator in the reactor-turbine thermodynamic cycle. Since, for a given emissivity, the required radiator area varies as the inverse fourth power of the absolute temperature, it is desirable to operate the radiator at as high a temperature as possible in order to keep the radiator mass small. However, a high radiator temperature means a small temperature difference across the turbine, and hence a small thermodynamic efficiency. An optimum radiator temperature exists at which the radiator mass, for a given power output of the turbine, is a minimum. In general, this optimum temperature is much higher than the heat sink temperatures of stationary powerplants. The turbine inlet temperatures must be as high as possible. At the present time, inlet temperatures of the order of 800°C are possible. It is interesting to note that the thermodynamic efficiencies resulting from this optimization procedure are of the order of only 10 to 15 per cent.

Another attractive type of system, particularly for low-power applications, would use solar batteries as a power source. At the present, they are too heavy and too expensive for space flight propulsion applications. However, if these problems can be overcome, their inherently high reliability and freedom from radioactivity, as well as from moving parts, would render their use possible at least in small propulsion systems.

Other types of solar power systems would use a lightweight parabolic or spherical mirror in conjunction with a thermionic or turbogenerator power conversion system.

The ionic propulsion unit itself would employ a contact-ionization type of ion source (see Sections IV and V for a more detailed description of the ionic propulsion unit). In such a source, the vapor of an alkali metal such as cesium or rubidium is passed over incandescent tungsten or platinum, both of which have higher work functions than the ionization potentials of cesium or rubidium. If the temperature of the tungsten or platinum catalyst is greater than a certain critical temperature (1200 to 1500°K), practically 100 per cent of all the alkali atoms which strike the hot catalyst will lose their

valence electrons to the catalyst and will evaporate as ions. The advantages of this kind of ion source are its long life, reliability, simplicity, and the large fraction of atoms which are ionized. The latter feature is necessary in order to efficiently utilize reaction mass.

One of the most serious problems in the development of a practical ionic propulsion unit lies in the space charge limitation upon the maximum current density which can be obtained with a given propellant material, accelerating voltage, electrode spacing and electrode geometry. The value of this maximum current density for a parallel-plate electrode geometry is given by the Schottky-Langmuir expression:

$$j = \frac{4}{9} \epsilon_0 \sqrt{\frac{2e}{m}} \frac{V^{3/2}}{x^2}. \qquad (1)$$

The ion beam would be accelerated by an annular cathode (Fig. 2). Focusing would be necessary to overcome divergence of the beam arising from the mutual repulsion of the positively-charged ions. The ion trajectories would have to be such that no appreciable fraction of the ions (<1 per cent) would strike the accelerating cathode. Otherwise, secondary electron emission, impact heating, and particularly electrode erosion would occur.

It is necessary to eject a separate stream of electrons at the same current level as the ion beam, first, in order to neutralize the positive space charge behind the ship; and second, in order to preserve the electrical neutrality of the ship.

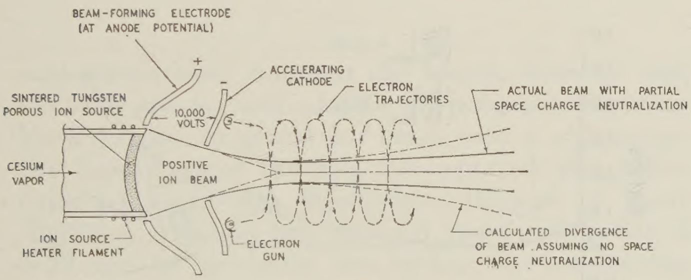

Fig. 2—Pierce gun type of ion accelerator. Path of electron trajectories through the axis of the beam is shown in dotted lines.

III. Design Features

The over-all design features of ionic propulsion systems will be discussed in limited detail. Since these systems may be optimized with respect to a number of criteria, the problem of optimum system design is a complicated one and depends considerably upon the mission which the system is to carry out.

In optimizing ionic propulsion systems with respect to various criteria, it will be assumed that the mass of the electric powerplant is directly proportional to its power output. However, since the power-to-weight ratio of a powerplant is approximately constant only over limited ranges of powerplant size, different values of power-to-weight ratio will be used for powerplants of greatly different capacity. It will also be assumed that the mass of the ionic thrust unit is directly proportional to its area.

For an interplanetary space ship, such as a Mars ship, it is of value to try to maximize the vehicle's initial acceleration. This is useful because a certain minimum acceleration is required in order to keep the time of spiralling around the earth reasonably short. The initial acceleration, a_i, may be expressed as a function of the propellant mass, M_F, for any given set of values for M_L, L, τ, e/m_i, α, and β, where:

$$a_i = \frac{\sqrt{\dfrac{2M_F L}{\tau}}}{M_F + M_L + \dfrac{L}{\alpha} + \beta\sqrt{\left(\dfrac{M_F e}{m_i \tau}\right)^5 \dfrac{1}{L^3}}}. \qquad (2)$$

Another type of optimization consists of minimizing the total initial-mass-to-payload-mass ratio required to accelerate a specified payload for a given period of time with a given initial acceleration. This is of interest in minimizing the total initial mass of a space ship which is to carry out an interplanetary mission. Here, the given parameters are τ, e/m_i, a_i, α and β. The total initial-mass-to-payload-mass ratio, M_0/M_L, may be expressed as a function of the exhaust velocity v_{ex}:

$$\frac{M_0}{M_L} = \frac{1}{1 - \dfrac{a_i \tau}{v_{\text{ex}}} - \left(\dfrac{a_i}{2\alpha}\right) v_{\text{ex}} - \dfrac{\beta a_i}{v_{\text{ex}}{}^4}}. \qquad (3)$$

A third type of optimization consists of minimizing the mass of propellant which must be transported into orbit to refuel a space ship. This will be of interest if a space ship is to repeatedly carry out a particular mission. The given parameters are M_L, τ, e/m_i, a_i, α and β. The ratio of propellant-mass-to-payload-mass M_F/M_L, is given by:

$$\frac{M_F}{M_L} = \frac{\dfrac{a_i \tau}{v_{\text{ex}}}}{1 - \dfrac{a_i \tau}{v_{\text{ex}}} - \left(\dfrac{a_i}{2\alpha}\right) v_{\text{ex}} - \dfrac{\beta a_i}{v_{\text{ex}}{}^4}}. \qquad (4)$$

A graph showing the effect of varying the exhaust velocity upon the initial gross-mass-to-payload-mass ratio of the Mars ship of Fig. 1 is shown in Fig. 3. The ordinate in Fig. 3 is given in terms of the ratio:

$$\mu_0 = \left(\frac{M_0}{M_L}\right) \bigg/ \left(\frac{M_0}{M_L}\right)_{\text{minimum}}.$$

Thus, it represents the dimensionless ratio of M_0/M_L to its optimum (minimum) value. The graph of Fig. 3 also

shows the ratios

$$\mu_F = \left(\frac{M_F}{M_L}\right) \Big/ \left(\frac{M_F}{M_L}\right)_{\text{minimum}}.$$

and

$$n_i = \frac{a_i}{(a_i)_{\text{ maximum}}}$$

as a function of the exhaust velocity for the Mars ship of Fig. 1.

Graphs showing μ_0 and μ_F for a perturbation correction propulsion system and for a satellite propulsion system are shown in Figs. 4 and 5, respectively.

Since the acceleration, initial mass and propellant mass curves of Figs. 3–5 have broad flat maxima and minima, it is possible to select an exhaust velocity which insures that all of the above quantities are reasonably close to their optimum values.

IV. Ion Source

One area of consideration in the development of ionic propulsion systems is the development of the ion source. Of the various types of conventional ion sources, the contact type seems the most promising because of its simplicity, high ionization efficiency,[1] and relative freedom from electrode erosion. The ratio of positive ions to impinging neutrals for a hot catalyst contact source is given approximately by the Saha-Langmuir equation [12]:

$$\frac{n_+}{n} + \frac{p_+}{p} \exp\left[= \frac{e(\phi - I)}{kT}\right]. \tag{5}$$

An experimentally-derived curve [12] showing the ionization efficiencies for various alkali metals vs tungsten catalyst temperatures is shown in Fig. 6. It may be seen that for temperature values somewhat greater than a certain critical temperature, the ionization efficiency for cesium is practically 100 per cent. The rapid falloff in ionization efficiency at temperatures below the critical temperature may arise because a layer of alkali metal is formed on the surface of the tungsten, lowering its work function (Table II). It is also of interest that an increase in the current density drawn from the source leads to an increase in the value of the critical temperature and may lead to a decrease in the ionization efficiency of the source. The variation of the critical temperature with current density is given approximately for cesium by the empirical equation [13]

$$T_c = \frac{14{,}350}{8.99 - \log_{10 j}}. \tag{6}$$

[1] In this paper, the term "ionization efficiency" will refer to the ratio of ions produced to the neutral atoms impinging upon the hot surface.

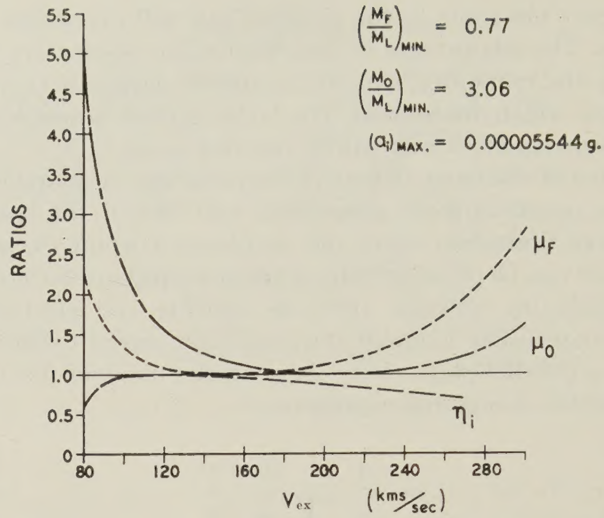

Fig. 3—Plot of μ_0, μ_F and η as a function of exhaust velocity, v_{ex}, for an ionically-propelled Mars ship.

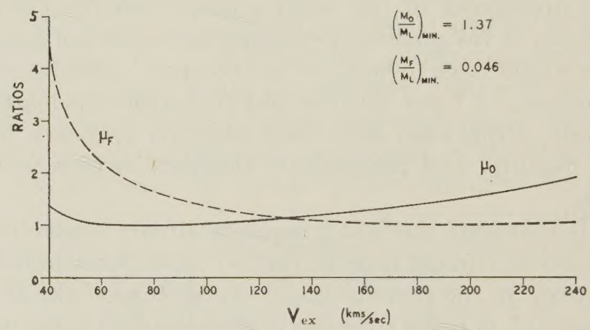

Fig. 4—Plot of μ_0 and μ_F as a function of exhaust velocity, v_{ex}, for a satellite perturbation-correlation ionic propulsion system.

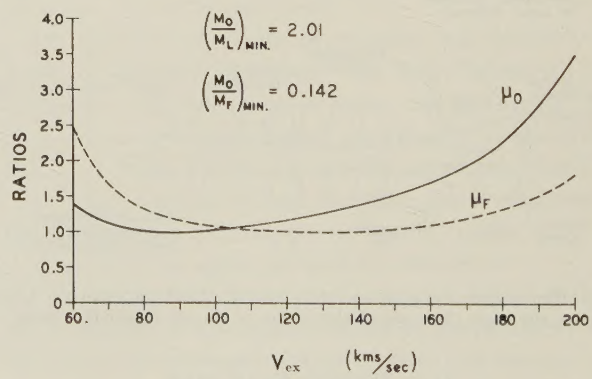

Fig. 5—Plot of μ_0 and μ_F vs exhaust velocity, v_{ex}, for a satellite propulsion system.

The flow rates and ionization efficiencies which can be obtained with a contact source depend upon the image forces exerted upon the ions, the magnitude of the applied electric field (if the current is space-charge-limited), and upon the thermal kinetic energy of the ions at the tungsten surface.

Experimental values of potassium-ion current density as high as 240 ma/cm^2 have been reported by the Thompson-Ramo-Wooldridge Corporation [14]. These current densities were obtained with a cylindrical geometry, a

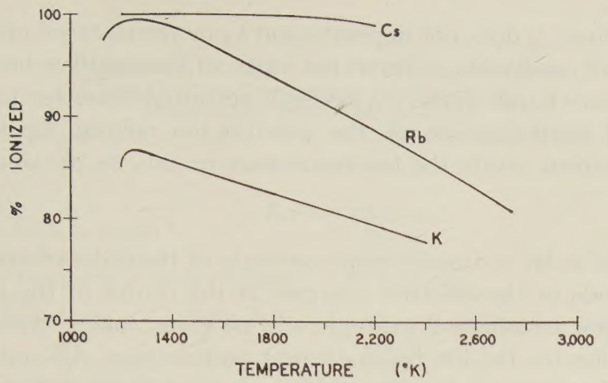

Fig. 6—Graph showing the fractional ionization of three alkali metals on a hot tungsten surface as a function of its temperature.

TABLE II
IONIZATION POTENTIALS OF THE ALKALI METALS AND WORK FUNCTIONS OF PT AND W IONIZATION CATALYSTS

Alkali Metal	Ionization Potential[16] (volts)
Li	5.36
Na	5.12
K	4.32
Rb	4.16
Cs	3.86
Contact Catalyst	**Work Function (volts)**
W(1600°K)	4.65[12]
W(2500°K)	4.72[12]
Pt	5.5[12]
W—O	6.51[12]
W—Th	2.63
W—Cs	1.6
W—O—Cs	0.71
W—Ba	1.56
W—O—Ba	1.34

potassium-coated tungsten ion source, close (1 mm) electrode spacing and an 800-volt potential difference. However, for an actual ion beam with a continuous-flow type of source, the maximum reported current densities are lower. The Rocketdyne Division of North American Aviation has reported current densities of 3–20 ma/cm^2 for cesium ions with a 6-kv accelerating voltage and an interelectrode spacing ranging from about 3 mm down to a fraction of a millimeter [15].

V. Space-Charge Neutralization

The focusing and acceleration of ion beams represents another important problem area in the development of ionic thrust devices. Focusing of the ion beam must be quite excellent in order to prevent the beam from striking the cathode accelerator electrode. Otherwise, secondary electron emission and rapid erosion of the cathode resulting from ion bombardment occur.

It is well known that space-charge-limited currents in electron beams can be somewhat larger than the values predicted by the Schottky-Langmuir Law, because of the production of positive ions in the beam by electron bombardment of residual gas molecules in the apparatus. Since the positive ion charge densities are much higher than electron charge densities for the same current density and accelerating voltage:

$$\left(\rho = \frac{j}{v_{ex}} = j\sqrt{\frac{m}{2eV}}\right),$$

positive ions are much more effective in reducing space charge in electron beams than are electrons in reducing space-charge effects in ion beams. However, space-charge effects in space-charge-limited ion beams can be alleviated somewhat by increasing the mass-to-charge ratio of the particles. For a given exhaust velocity, the thrust-per-unit-area which can be obtained with a space-charge-limited beam increases with particle mass according to the relation

$$P = \frac{m_i^2 v_{ex}^3}{2\sqrt{2}e^2x^2}. \tag{7}$$

If a constant exhaust velocity is to be maintained, the accelerating voltage must be increased in direct proportion to the mass-to-charge ratio of the ionic propellant. Consequently, a practical upper limit to the permissible mass-to-charge ratio is imposed by the value of the breakdown voltage gradient.

Since a collimated ion beam is desirable to obtain maximum thrust, a cylindrical-beam or converging-beam Pierce Gun accelerating system may be suitable, as shown in Fig. 2.

Another serious problem is that of neutralizing the ion beam. In the configuration shown in Fig. 2, the electrons are injected into the ion beam from electron emitters spaced evenly around the perimeter of the beam. Ideally, the electrons would be injected into the beam with the same vector velocity as the positive ions. However, the electrons' random thermal emission velocities alone are comparable in magnitude to the velocity of the accelerated ion beam. Furthermore, the electrons are drawn into the ion beam because of the positive space charge in the unneutralized portion of the beam. In these positive space-charge fields, the electrons acquire radial velocities which are an order of magnitude greater than their random thermal velocities. Since the particle densities in the ion beam are low (about 10^{10} particle/cm^3), the probabilities of the occurrence of radiative and three-body recombination, and collision ionization, are low for an ion beam of moderate diameter, and the electrons pass through the beam with essentially no loss in energy. Consequently, they travel through the beam and out to a radial distance equal to their injection radius as shown in Fig. 2.

It is difficult to obtain an exact solution for the trajectories of the charged particles in these fields since the fields depend upon the paths of the particles, and the paths of the particles depend upon the disposition of the fields. In practice, a self-consistent field method leading

to a solution by successive approximations may be employed. As a first simple approximation to the actual potential and particle distribution, the positive ion beam will be treated as a cylindrical column of uniformly-distributed positive charge of radius a and of infinite length into which the electrons are injected. The justification for this approximation is that such a model permits a theoretical approach to a problem which, under real physical conditions, is too complex for an analytical treatment in closed form. The kinetic energy of an electron accelerated from rest at an initial radial distance r_0 from the axis of the infinite cylinder of positive charge is

$$W = \frac{1}{2} m_e \dot{r}^2 = \frac{\lambda_+ e}{2\pi\epsilon_0} \ln \frac{r_0}{r} \qquad \text{for } r \geq a; \tag{8a}$$

$$W = \frac{1}{2} m_e \dot{r}^2 = \frac{\lambda_+ e}{8\pi\epsilon_0} \left[4 \ln \frac{r_0}{a} + \left(1 - \frac{r^2}{a^2}\right) \right] \qquad \text{for } r \leq a. \tag{8b}$$

Eqs. (8a) and (8b) may be solved for $\dot{r}$ and integrated to obtain t (time) as a function of r. This yields:

$$t = r_0 \left(\frac{4\pi\epsilon_0 m_e}{\lambda_+ e} \right)^{1/2} \int_0^{\sqrt{\ln r_0/r}} e^{-s^2} ds \qquad \text{for } r \geq a \tag{9a}$$

$$= r_0 \left(\frac{4\pi\epsilon_0 m_e}{\lambda_+ e} \right)^{1/2} \operatorname{erf} \left(\ln \frac{r_0}{r} \right)^{1/2}$$

$$t = a \left(\frac{4\pi\epsilon_0 m_e}{\lambda_+ e} \right)^{-1/2} \left[\sin^{-1} \left(4 \ln \frac{r_0}{a} + 1 \right)^{-1/2} + \frac{r_0}{a} \operatorname{erf} \left(\ln \frac{r_0}{a} \right)^{1/2} \right] \qquad \text{for } r \leq a. \tag{9b}$$

The above equations, which apply only for the first quarter period of oscillation, may readily be generalized to successive oscillations. However, since the electrons lose essentially no energy in traversing the beam, the above expressions (9a) and (9b) are sufficient to define the equations of motion. It may be seen from the potential function in (8b) that the electrons execute simple harmonic motions within the beam. It is also clear that the maximum displacement of the electrons, if they start from rest and lose no energy in the beam, will be equal to their original displacement.

It is of interest to determine the fraction of the time an electron spends within the positive ion beam. If

$$R = \frac{r_0}{a} \text{ and } \tau_b = \frac{t_b}{T}$$

where τ_b is the time the electron spends in the beam during a single oscillation and T is the period of oscillation, then:

$$\tau_b = \frac{\sin^{-1} (1 + 4 \ln R)^{-1/2}}{\sin^{-1} (1 + 4 \ln R)^{-1/2} + \rho \operatorname{erf} (\ln R)^{1/2}}. \tag{10}$$

Since τ_b does not depend upon λ_+ or the electron energy of oscillation, a corrected value of the positive beam linear-charge density λ_+' which accommodates for partial neutralization of the positive-ion column by the electrons inside the ion beam may readily be obtained:

$$\lambda_+' = \tau_b \lambda_+. \tag{11}$$

In order to furnish some estimate of the order of magnitude of the electron energies at the center of the ion beam, a numerical example will be given, using typical values for the ion beam current and voltage. Assuming an ion current density of 10 ma/cm^2, a beam radius of 0.5 cm, electron emitters located on a circle of 0.75-cm radius, and an ionic velocity of 100 km sec^{-1} (corresponding to an accelerating voltage of 6.8 kv for cesium ions), the kinetic energy of an electron at the center of the beam will be 3700 electron volts. The fraction of the time the electrons spend within the beam, τ_b, is equal to 0.523. Therefore, the corrected energy of electrons at the center of the beam, taking into account the neutralization effect of the electron transit time through the beam, is approximately 2000 electron volts.

The model described above is, of course, greatly over-simplified. It neglects electronic space charge effects and the shielding of electrons at large radii by electrons at smaller radii. The latter effect tends to decrease the potential of the electrons at the center of the beam. The model also assumes a uniform cylindrical distribution of positive ionic charge. In practice, the positive space-charge distribution will be nonuniform, finite, and will diverge laterally with increasing distance from the cathode because of the mutual repulsion of the positive ions. However, the simplified treatment given here indicates the oscillatory behavior of electrons injected into the ion beam, and gives some order of magnitude indication of electron energies and of the degree of fractional neutralization in the ion beam.

VI. Applications

Four prominent applications suggest themselves for an ionic propulsion system. First, it could be used to propel interplanetary space ships. In this application, it would furnish an order of magnitude improvement in payload-to-total mass ratio over a chemically-powered rocket. Clearly, this feature is of great importance if propellants must be transported from the earth and placed in a satellite orbit.

Second, it could be used to correct perturbations in the orbits of earth satellites. This would be particularly useful in correcting the "drift" tendencies of a communications relay satellite. In this application, a definite advantage would be the precise control and metering of thrust which it would afford. Also, an ionic propulsion system will provide a greater total velocity change than a chemically-powered system of the same weight. A typical set of parameters for an orbital correction ionic propulsion system is given in Table III.

TABLE III

Electrostatic Satellite Propulsion System for Orbital Correction Applications; Design and Performance Data

Assumed Parameters	
Total weight	10,000 lb
Powerplant weight	1,950 lb
Propellant weight	550 lb
Power-to-weight ratio of powerplant	10 watts/kg
Specific mass of ionic thrust unit	3.64×10^{3} metric tons volts$^{3/2}$ amps^{-1}
Power utilization efficiency of ionic propulsion unit	0.68
Thrust	0.05 lb
Initial acceleration	5×10^{-6} g
Time of operation	Approximately 3 yrs
Electrode spacing	0.01 meter
Calculated Parameters	
Specific impulse	8,160 sec
Accelerating voltage	4,230 volts
Total area of ion propulsion units	0.156 square meter
Characteristic velocity	4.41 km sec^{-1}
Total power requirement	8.91 kw

TABLE IV

Design and Performance Data for an Electrostatically Propelled Satellite

Assumed Parameters	
Total weight	20,000 lb
Powerplant weight	6,520 lb
Propellant weight	1,840 lb
Power-to-weight ratio of powerplant	30 watts/kg
Specific mass of ionic thrust unit	3.64×10^{3} metric tons volts$^{3/2}$ amps^{-1}
Power utilization efficiency of ionic propulsion unit	0.68
Thrust	0.5 lb
Initial acceleration	2.5×10^{-5} g
Time required to go from a low orbit to a 22,000 mile orbit	Approximately 1 year
Electrode spacing	0.01 meter
Calculated Parameters	
Specific impulse	8,160 sec
Accelerating voltage	4,230 volts
Total area of ionic propulsion unit	1.56 square meters
Characteristic velocity	8 km sec^{-1}
Total power requirement	89 kw

Third, it could be used to transport satellites from low orbits ($\sim$300-mile altitudes) to high orbits ($\sim$22,000-mile altitudes). The more effective utilization of propellant mass which ionic propulsion affords would permit the delivery of a larger fraction of the low altitude payload mass to a higher orbit than could be achieved with a chemical propulsion system. With the latter, only about one-third to one-fifth of the low altitude payload mass could be delivered as payload to the higher orbit. Also, once an ionically-propelled satellite were established in a higher orbit, only a small fraction of its electrical power generation capability would be required for perturbation corrections. Consequently, the major part of its electrical power output could be diverted to other purposes, such as that of powering a communications relay.

Finally, an ion rocket could be used to propel small, unmanned interplanetary space probes. Here again, at certain times, the ionic propulsion unit could be deactivated and its electrical power generation capability employed to transmit data to the earth.

A set of design parameters for satellite and space probe propulsion systems is given in Table IV.

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Plasma Propulsion Devices for Space Flight*

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Summary—An analysis of some of the more immediate space missions indicates that a large increase in payload can be achieved when electrical propulsion is used instead of chemical propulsion. For missions in the gravitational field of the Earth and to the Moon, the optimum specific impulse range for electrical propulsion is from about 1500 to 5000 seconds. Electrical propulsion with neutral plasma devices operate well in this specific impulse range as well as at higher specific impulses. Three different chambers have been described as examples of devices using neutral plasmas. Some of the factors which limit the range of efficient operation of such devices have been discussed.

I. Introduction

ELECTRICAL propulsion was suggested many years ago as an enticing possibility for reducing the cost of propelling large payloads in space. After a vehicle has been launched into a low satellite orbit by chemical rockets, high thrust levels are no longer required. The use of electrical propulsion from this point on would increase greatly the size of the payload which can be propelled into higher orbits. For many near space missions (see Fig. 1) the potential increase in payload may be two to three times greater if electrical propulsion is used.

In specific cases, such as placing a communication satellite in a 24-hour "stationary" orbit at 22,500 miles, there is an additional advantage in the use of electrical propulsion. This results from the fact that the power supply has a dual function in subsequently providing power for the communications system. It can be shown that for this communication satellite only one-third as much weight must be placed into low orbit if the transfer to the 22,500-mile orbit is made by electrical propulsion rather than by using chemical rockets throughout the trip. This means that an initial booster vehicle one-third as large could accomplish the same mission.

In the next section an analysis is made of the more immediate space missions in order to specify requirements for the propulsion chamber. It is shown that, for missions in the gravitational field of the Earth and to the Moon, the optimum specific impulses for electrical propulsion lie between 1500 and 5000 seconds. Operation in this range is inefficient for the ion rockets currently under development. However, thrust devices using a neutral plasma as the propellant gas can be made efficient in this specific impulse range. In Section III a discussion of neutral plasma devices will be given and the important limitations to their operation will be discussed.

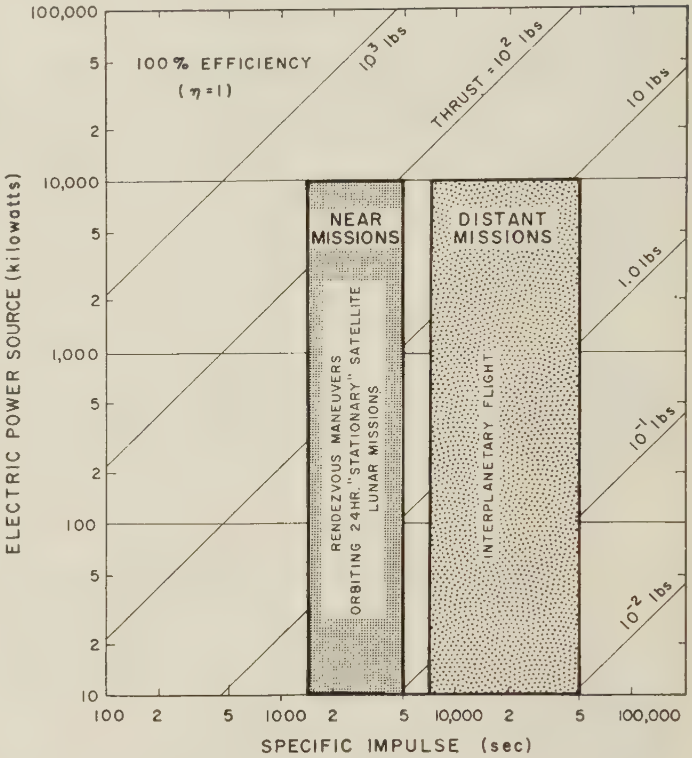

Fig. 1—Area of interest for electrical propulsion. The electric power level is plotted as a function of specific impulse. Lines of constant thrust are shown. The shaded areas represent the optimum specific impulse range for representative missions.

II. Optimum Specific Impulse Ranges for Various Space Missions

Any discussion of the applicability of electrical or any other space propulsion device must necessarily be complemented by some form of mission analysis. We have attempted to determine the most desirable specific impulse ranges for which electrical propulsion systems should be developed. For this analysis the following missions were considered: 1) Placing a communication satellite in the 24-hour orbit. 2) A round trip to a lunar orbit. 3) Rendezvous maneuvers. 4) Interplanetary flights.

It was felt that the first three of these missions were representative of practical space ventures in the not too distant future. While the main emphasis of this analysis is the evaluation of electrical propulsion devices, a comparison is also made with chemical rockets and solar energy converters. The results of this analysis as they pertain to electrical propulsion systems can be summarized as follows:

For missions within the gravitational field of the Earth, such as 1) and 2) above, the optimum specific

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impulse range is from 1500 to 5000 seconds. For a mission such as 1) above, the communication satellite, electrical propulsion should receive particular attention because the power supply can also be used for the communication system. For interplanetary flights, the desirable specific impulse range is from 7500 to over 20,000 seconds. Fig. 1 is an electric power level vs specific impulse graph showing the range of operation for electrical propulsion.

Before giving numerical examples for various missions, we will discuss briefly some general features of electrical propulsion. The optimization of flight parameters for electrical propulsion differs in some respects from chemical rocket propulsion, since the source of power and the working fluid are separate items. For example, in chemical propulsion there is a premium on obtaining the highest specific impulse because the thrust power is limited to the specific energy storage within the propellant. On the other hand, practically any value specific impulse is attainable with electrical propulsion devices. However, the optimum specific impulse is *not* the highest one. This becomes clear when one considers the dependence of the propellant and power plant weights on the specific impulse. The propellant weight decreases with increasing specific impulse. On the other hand, the power plant weight increases with increasing specific impulse. The optimum performance is obtained when the payload weight is maximized, or, conversely, when the combined weight of the propellant and power plant is minimized. The optimum payload weight is obtained when the power plant weight equals the propellent weight.

Power Supply Weight and Thrust Efficiency

In a general mission analysis, the entire electrical propulsion system is characterized by its weight and thrust power output. However, only the ratio α/η appears in the formulas. The power plant specific weight, α, is the ratio of the power plant weight to electric power output; and the efficiency of the thrust chamber, η, is the ratio of the power in the exhaust jet to the power supplied the propulsion unit. In most analyses a value for η of unity is tacitly assumed. However, it is important to remember that any inefficiency of the thrust chamber will require an additional power supply weight in order to perform a given mission.

The required weight of a propulsion system is inversely proportional to the product of the efficiencies for a) the generation of electric power, and b) the conversion of electric power into directed thrust motion. Consequently, the thrust chamber and power generation efficiencies have equal importance in determining the weight of the power supply.

Communication Satellite

As the first example, we will consider the problem of establishing a 24-hour "stationary" communication satellite in orbit. This mission was considered because it is an example of the type of mission for which the electrical power requirements in orbit are very large and consequently, the power supply for the electrical propulsion device is essentially free. There are two parts to this problem: first, placing the satellite in orbit; and second, maintaining the satellite at a stationary point above the Earth for a long time interval. Fortunately, the orbit perturbations, due to the Earth's bulge, the Sun, and the Moon, are small enough so that no orbit corrections would be required for the first year.[1]

Table I gives a comparison of several propulsion systems for bringing a 5000-pound communication satellite from a 150-mile circular orbit to the 24-hour orbit (approximately 22,000 miles altitude). Three propulsion systems are considered; the chemical rocket, hydrogen heater,[2] and electrical propulsion. Different electric propulsion thrust devices are listed separately as they operate over widely different specific impulse ranges. Representative working fluids are given in the second column. The third column lists the specific impulses that appear feasible for these thrust devices. The payload weight ratio (the payload weight to the initial weight at 150 miles) is shown in the next column. The weight of the propellant drops rather rapidly as the specific impulse increases. The initial weight requirement at 150 miles is computed by adding the 5000-pound payload to the weight of the propellant plus tankage. Note that the required initial weight drops by a factor of three from about 18,000 to 6000 lbs if electrical propulsion is used instead of chemical rockets. This factor of three in the launching weight requirements is a most attractive feature of electrical propulsion.

The last two columns show the flight time and assumptions used to compute this time. For chemical propulsion, the 4.0 hours flight time is the coasting time for a minimum energy orbit (the Hohmann transfer ellipse). For the hydrogen heater and electrical propulsion systems the flight time depends on the power available. In the case of the hydrogen heater it was assumed that solar energy was collected by a 60-foot diameter collector and used with 50 per cent efficiency to produce thrust. For the electrical propulsion systems the specific weight of the power supply divided by the thrust chamber efficiency α/η, was taken as 50 lbs per kw. The value of the specific weight was based on the use of available solar batteries. Further, it was assumed that the power supply weight was one-half of the payload weight. The assumptions resulted in 50 kw of thrust power. Any variation of the fraction of the payload allotted to the power supply would change the flight time proportionately but would not affect the payload weight ratio.

It should be noted that the electric power supply, ex-

[1] D. Lautman, "Perturbations of a Twenty-Four Hour Satellite Orbit," Avco Res. Lab., RR-43; 1959.

[2] A solar-powered hydrogen rocket, limited by the maximum temperature which can be contained in the "boiler."

TABLE I

24-Hour Orbit Communication Satellite with 5000-lb Payload

Propulsion System	Propellant Material	Specific Impulse	Payload Weight Ratio	Propellant Weight plus 10 Per Cent Tankage	Initial Weight	Flight Time, T	
Chemical	H_2–O_2	350 seconds	0.267	13,750 lb	18,800 lb	4.0 hrs	Hohmann transfer ellipse two-stage rocket.
		400 seconds	0.318	10,720	15,700	4.0 hrs	
H_2 heater	H_2	600	0.400	7,500	12,500	2.1 days	60-foot diameter solar collector with 50 per cent conversion efficiency.
		700	0.455	5,960	11,000	2.5	
		800	0.502	4,950	10,000	2.9	
Electrical: Arc Heater	He	1,100	0.602	3,300	8,300	19 days	Power supply included in payload and is one-half of the payload weight, $\alpha/\eta=50$ lbs per kw, *i.e.*, 50-kw thrust power.
		1,500	0.692	2,200	7,200	23	
MHD	Li	2,000	0.765	1,540	6,540	28	
		5,000	0.898	570	5,570	64	
		10,000	0.948	280	5,280	131	
Ion rocket	Cs	7,500	0.929	380	5,380	98	
		10,000	0.948	280	5,280	131	

cept for the thrust chamber and guidance, was considered as part of the payload, since it can be used both for propulsion into orbit and subsequently for the power for the communication electronics. A case where the power supply can be varied and is not part of the payload will be considered for the Moon mission. For the hydrogen heater no weight penalty was allowed for the radiation collection dish. This was done under the assumption that the collection dish would also be used as an antenna and therefore can be considered as part of the payload.

From the data given in Table I we have plotted in Fig. 2 the flight time vs the payload weight ratio. Also shown are specific impulse ranges for the various thrust devices. Note that the flight time increases rapidly when the specific impulse is above about 3500 seconds with only small further increases in the payload weight ratio. Thus, the optimum specific impulse for this mission would appear to be in the range from about 1500 to 5000 seconds, depending on the flight time required.

Moon Mission

For a Moon mission we have assumed that a large power supply is not required, and that the power supply cannot be considered as part of the payload. Thus, one is at liberty to choose the power plant size that optimizes other mission parameters. Fig. 3 (opposite) showed the result of an analysis for a round trip mission to the Moon. The payload weight ratio is given as a function of the specific impulse for several flight times. The velocity increment Δv is about 500 seconds for a round trip to a lunar orbit starting from 150 miles circular orbit.

To obtain these curves a power supply specific weight of 40 lbs per kw was assumed, of which 26 lbs per kw was used for batteries and 10 lbs per kw for the rest of the power supply; the efficiency of the propulsion chamber $\eta=0.9$. The 10 lbs per kw is a somewhat futuristic assumption for the specific weight of a power supply. For a power supply specific mass of 50 lbs per kw then the

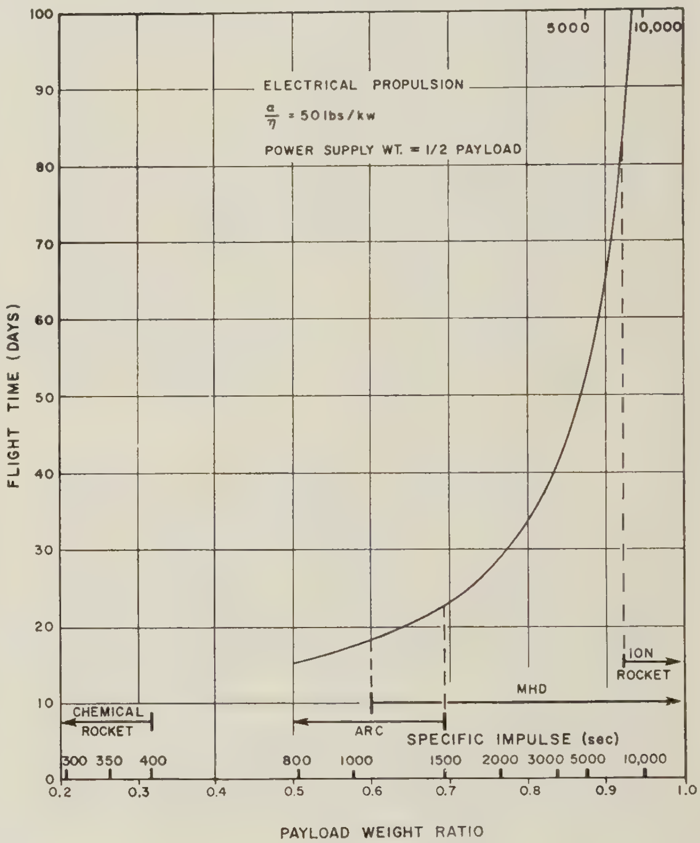

Fig. 2—Comparison of several propulsion systems for placing a communication satellite into a "stationary" 24-hour orbit starting from a circular 150-mile orbit. The electric power supply is included in the payload since it is also used to power the communications equipment.

curves would remain the same if the flight times are increased by a factor of 1.5.

For each flight time there is shown to be an optimum specific impulse for which the payload weight ratio has a maximum value. The payload weight is less for lower specific impulses than this optimum because more propellant weight is needed; while for larger specific im-

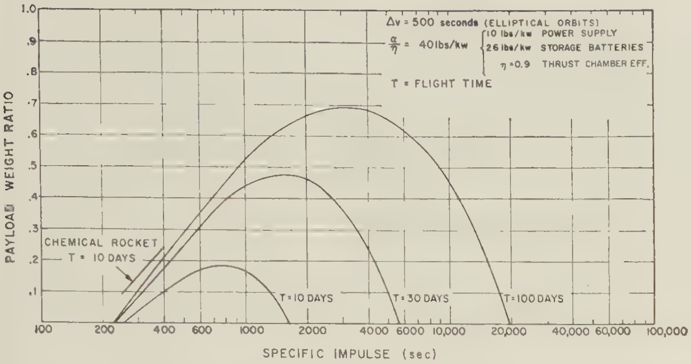

Fig. 3—Payload capability for a round trip to a lunar orbit as a function of specific impulse and flight time. Flight accomplished in optimum manner by elliptical orbits and use of energy storage as discussed in Table I.

pulses the power plant weight has increased too much in order to maintain the necessary thrust level.

The flight time for this mission for chemical propulsion is ten days with the minimum energy path. Shorter flight times can be achieved, but only with a reduction in the payload weight ratio. Note that when using electrical propulsion the initial weight requirement at 150 miles is reduced by a factor of 2 for a 30-day flight time. For a 100-day trip, then there is a factor of 3 reduction over the initial weight requirement for chemical rockets. Thus it can be seen that the use of electrical propulsion can effect economies of the order of 2 or 3 in booster costs.

Rendezvous Maneuvers

Since the types of rendezvous missions which may be of interest will vary in the Δv requirements, as well as in the time allotted for a particular maneuver, a very wide range of specific impulse appears to be useful. Also a number of individual maneuvers can easily be anticipated for a single vehicle. Consequently, the range of specific impulse of interest for this purpose can range from values attainable with chemical rockets to possibly as high as 10,000 seconds.

Interplanetary Trips

Detailed mission analysis for a Mars trip was made by Irving and Blum,[3] and they indicated that specific impulses in excess of 10,000 seconds are needed.

Summary

We have attempted to show that flights starting from a low altitude circular orbit to larger orbits around the Earth or to the Moon require specific impulses in the range from 1500 to 5000 seconds. For some applications, where the payload must include an electrical power plant, electrical propulsion appears to be particularly attractive. For interplanetary flights, specific impulses in excess of 10,000 seconds are desirable. Due to the large variety of possible missions electrical propulsion devices over a large range of specific impulses appear to be required. However, for the more immediate military missions such as the "stationary" communication satellite and Moon missions, the desirable specific impulse range is from 1500 to 5000 seconds. For these missions electrical propulsion can effect a reduction in initial ground level launch weight requirements by a factor of two or three; conversely, for the same ground rockets, a two to three times larger payload can be delivered to the final orbit by electrical propulsion.

III. Operating Ranges of Various Thrust Devices

It is convenient to separate thrust devices into three categories based upon the methods of transferring energy to the propellant. These three categories are: 1) nonconducting gas devices (chemical rockets, nuclear rockets, solar heaters); 2) ion beam devices (ion rockets); and 3) neutral plasma devices. The last two of these are electrical propulsion devices, while the first one is not. A brief description of the devices in these categories will be presented together with their specific impulse ranges. Following this is a discussion of the basic design and operation of neutral plasma devices.

1) In nonconducting gas devices, energy is either stored in the working fluid in the form of chemical energy, or energy is transferred to the fluid by contact with hot chamber walls. In both of these cases the maximum specific impulse attainable is limited to rather low values. Chemical rockets will probably not exceed specific impulses of 400 seconds in the near future.

For devices in which the propellant is heated by contact with walls, the limiting stagnation temperature is determined by the maximum practical temperature of the wall material. Two systems which utilize the wall heating principle are a) the nuclear rocket (Project Rover) in which the hydrogen is heated by flowing through a nuclear reactor, and b) the solar heater in which a mirror is used to focus sunlight on a boiler or part of the thrust chamber. The problem of containing the gas has been considered,[4] and it is seen that at a specific impulse of 750 seconds the gas temperature, and hence the containing wall temperature, must be 4000°F (2500°K). Such a temperature is near the limit of the capability of existing materials.

2) The most widely known electrical propulsion system is the ion rocket which employs electrostatic acceleration of ion beams. Atomic ions are formed and accelerated to the desired velocity in an electric field. In order

[3] J. H. Irving and E. K. Blum, "Comparative Performance of Ballistic and Low-Thrust Vehicles for Flight to Mars," Second Annual AFOSR Astronautics Symp., Denver, Colo.; April, 1958.

[4] K. A. Ehricke, "Comparisons of Propulation Systems: Solar Heating, Arc-Thermodynamics and Arc-Magnetohydrodynamic," presented at Advanced Propulsion Systems Symp., Los Angeles, Calif.; December, 1957.

to exert a net force, the accelerating field must act on a region where only one type of charge is present. Therefore, the ions and electrons must be separated and only the ions allowed to flow through the accelerating field.

The principles of the ion rocket are well understood, and it is clear that it will be difficult to attain efficient operation at low specific impulses. This lower limit is determined by practical considerations based upon several fundamental effects: the ion space-charge current limitation in the accelerating region, electrode erosion by heavy ion bombardment, and radiation from the hot surfaces.[5,6] The practical lower limit to the specific impulse which can be achieved with ion rockets appears to be about 7500 seconds.[5] A more detailed discussion of the ion rocket is presented in other papers of this issue.

3) In neutral plasma devices, the propellant is a partially or fully ionized gas. The electron and ion densities are maintained essentially equal so that there is no net space charge in the gas and consequently the space charge limitations associated with the ion rocket are avoided. Since the gas is an electrical conductor, it will interact with electromagnetic fields. Electromagnetic fields can be used either to heat the gas to high temperatures or to accelerate it to high velocities. The gas stagnation temperatures which can be achieved do not depend on the chamber wall temperature as energy is transferred directly to the body of the gas. Specific impulses greatly in excess of those which can be reached with nonconducting gas devices are therefore attainable. In principle, neutral plasma devices can be conceived that operate over a range of specific impulses from *below 1000 to above 20,000 seconds*. Neutral plasma devices may therefore be expected to bridge the gap between the nonconducting gas devices and the ion beam, and to overlap the operating ranges of these devices to some extent. In particular, they can operate in the important specific impulse range which is most favorable for flight in the field of the Earth and to the Moon.

Neutral Plasma Thrust Chambers

A neutral plasma is an electric conductor and can be acted upon by electric and magnetic fields. Energy can be transmitted to a plasma electromagnetically in two basic ways:

1) A current in the plasma produces ohmic (Joule) heating thus increasing the thermal energy of the gas.
2) A plasma current flowing in the presence of a magnetic field produces body forces which accelerate the plasma and can be used to increase the energy of directed motion of the gas.

There are several ways in which magnetohydrodynamic forces can act upon a plasma. Depending upon the plasma electrical conductivity, the geometry of the apparatus and the time interval of interaction of the plasma with the electromagnetic fields, electric currents will flow either on the surface of the plasma or will be distributed throughout the body of the gas. The resulting forces on the plasma may be either on the boundary of the gas, as in the flow of nonconducting gases, or may be distributed throughout the volume. Furthermore, since magnetic field lines resist bending, a plasma containing a magnetic field possesses a rigidity along the magnetic field lines.

Due to the variety of ways in which magnetohydrodynamic forces can be made to act on a plasma and the variety of ways in which these forces can be transmitted between different parts of the plasma, a large number of different devices for accelerating neutral plasmas are possible. The operation of three neutral plasma accelerating devices will now be briefly discussed.

The Arc Jet

The schematic drawing shown in Fig. 4 indicates the basic features of the arc jet. A dc electric voltage in the range from 100 to 300 volts is applied between the anode and cathode electrodes. Depending on the operating power level the current will range from less than 100 to more than 1000 amperes. The temperature of the propellant gas is increased by Joule heating as it flows through the electric arc. The gas then passes through a settling chamber where sufficient time is allowed for nonuniformities in the gas to be reduced. The hot gas is then expanded in a conventional nozzle. The upper limit on specific impulse to the efficient operation of the arc jet is determined by frozen flow losses.

Frozen Flow

When a gas is heated to temperatures above about 3000°K dissociation and ionization begin to occur. This means that some of the energy which has been used to heat the gas has been invested in separating the molecules into atoms or into detaching electrons from atoms. When the gas is cooled as it expands through a nozzle, this energy is lost and does not appear as kinetic energy in the exhaust unless recombination occurs before the gas leaves the nozzle. This effect is well known in chemical rockets where the energy absorbed by vibrational excitation and molecular dissociation in the combustion chamber is not completely recovered as the gas temperature drops in the expansion nozzle. Since the stagnation temperatures for plasma propulsion devices are higher than for chemical rockets a larger fraction of the energy will be invested in dissociation and ionization, and the resulting losses may be larger. As will be seen, these losses can, however, be avoided in certain devices.

Frozen flow losses are most serious for a device such as the arc jet where the gas is heated to the stagnation temperature. The efficiency which would be obtained, assuming no recombination in the nozzle and neglecting

[5] R. H. Boden, "The Ion Rocket Engine," presented at Advanced Propulsion Systems Symp., Los Angeles, Calif.; December, 1957.

[6] S. Naiditch, "Ion Propulsion Systems," presented at Advanced Propulsion Systems Symp., Los Angeles, Calif.; December, 1957.

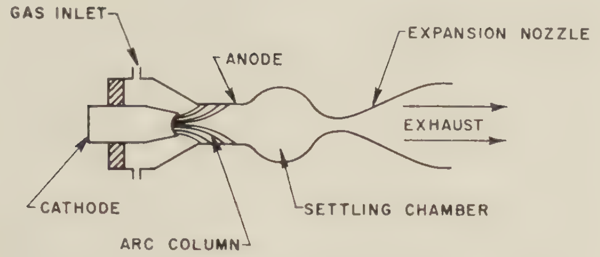

Fig. 4—Schematic diagram of an arc jet propulsion device.

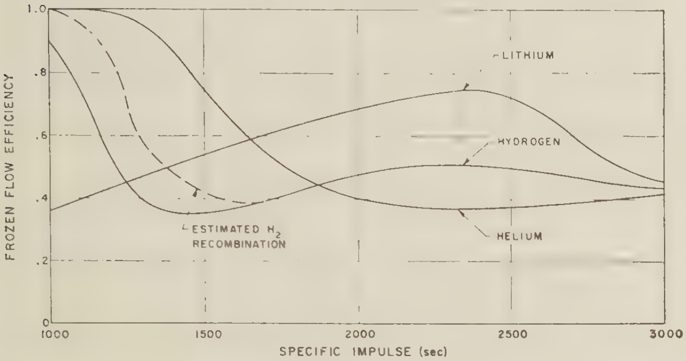

Fig. 5—Frozen-flow efficiency applicable to arc jet propulsion devices.

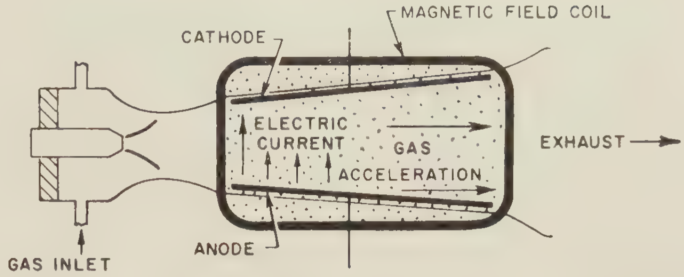

Fig. 6—Low temperature magnetohydrodynamic accelerator.

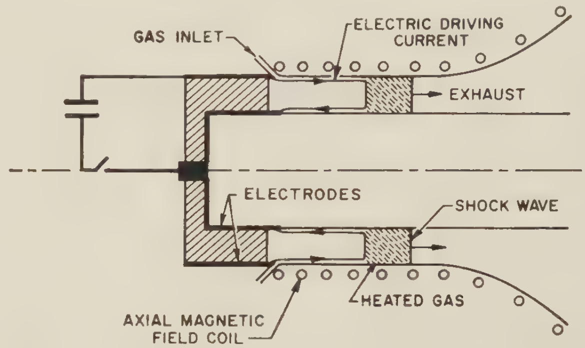

Fig. 7—Magnetohydrodynamic shock tube accelerator.

all other losses, is plotted in Fig. 5 as a function of specific impulse for several gases. The frozen flow efficiency is defined as the ratio of the thrust power to the power which is required to heat the gas to the stagnation temperature. Note that helium has the highest efficiency for specific impulses below about 1600 seconds and above this value lithium is best. Hydrogen has a lower efficiency due to losses in molecular dissociation.

If recombination occurred, higher efficiencies would result. For both helium and lithium best estimates indicate that for reasonable pressures and nozzle sizes recombination would not occur during expansion. For hydrogen the recombination rate is not well known. An estimate of the maximum improvement in efficiency due to recombination of hydrogen is indicated by the dash line in Fig. 4.

Low Temperature Magnetohydrodynamic Accelerator

A low temperature magnetohydrodynamic accelerator utilizes magnetohydrodynamic forces to accelerate a gas (see Fig. 6).[7] The propellant gas is preheated to a temperature of at least 3000°K by an electric current in an arc jet chamber similar to the one described in Fig. 4. The propellant then passes into the acceleration chamber. Currents are made to flow through the gas between the electrodes placed at the top and bottom of the duct. A magnetic field traverses the duct perpendicular to the plane of the figure. These currents in the presence of the magnetic field produce a force which accelerates the gas. The accelerating force is perpendicular to the magnetic field and to the electric current. The gas temperature need not be increased throughout the acceleration process. Consequently, frozen flow losses are minimized and this device can extend the specific inpulse range for the arc jet from 1500 seconds to about 5000 seconds.

Magnetohydrodynamic Shock Tube

A schematic drawing of this device is shown in Fig. 7.[8] Gas enters an annulus consisting of two concentric cylinders. This device operates on a pulsed basis. When the switch shown in the circuit diagram is closed, a voltage appears between the electrodes in the annulus gap, and there is a doughnut-shaped electrical discharge. As the capacitor discharges, the current loop in the gas will expand, pushing the gas in the annulus ahead of it and setting up a shock wave. The driving force is the magnetic field produced by the currents. This force acts on the current loop and tends to expand it.

The drawing shows the shape of the current loop after it has expanded down the tube. The coils surrounding the device produce a weak axial magnetic field which forms a "magnetic bottle" and inhibits plasma flow to the walls, thereby reducing energy losses to the walls. Note that the gas between the shock wave and the current loop is both heated and given directed motion. In this device about one-half of the energy goes into directed motion, and the other half into thermal energy. The gas expansion at the exhaust end of the chamber permits recovery of some of this thermal energy. The estimated

[7] R. J. Rosa, "Application of Magnetohydrodynamics to Propulsion," presented at Advanced Propulsion Systems Symp., Los Angeles, Calif.; December, 1957.

[8] R. M. Patrick, "A Description of a Propulsion Device which Employs a Magnetic Field as the Driving Force," presented at Second Annual AFOSR Astronautics Symp., Denver, Colo.; April, 1958.

specific impulse range of efficient operation of this device is from about 1000 seconds to over 20,000 seconds.

Electrodeless Devices

The three devices discussed above all require electrodes. This is not a necessary requirement. Several electrodeless devices have been suggested. It is, however, not clear that the advantage gained from eliminating the electrodes is not outweighed by the additional electronic circuitry problems which arise.

Energy Losses

The three major effects which may limit the range of efficient operation of neutral plasma devices are: 1) Losses due to frozen flow in the nozzle; 2) Heat losses to the chamber walls; and 3) Losses at electrode surfaces.

Frozen Flow Losses: These were considered previously in the arc jet. Such losses can be greatly reduced if the gas temperature is mtaintained lower than the stagnation temperature. This can be accomplished by the use of magnetohydrodynamic forces which can convert the electrical energy directly into kinetic energy of the gas, thus allowing the addition of energy to the gas without raising its temperature. Since the maxinum gas temperature is less than the stagnation temperature the energy invested in dissociation or ionization is lower and the frozen flow losses are therefore reduced.

Heated Losses to the Chamber Walls: One of the important effects that can lead to inefficient operation is the transfer of energy from the propellant gas to the chamber walls. In a nonconducting gas or in a plasma without magnetic fields, such losses occur by the conduction and diffusion of energy to the containing walls. In the case of a plasma with high electrical conductivity, these losses can be appreciably reduced by means of a magnetic field; *i.e.*, a magnetic field can be used to keep the plasma away from the solid walls. This is termed "magnetic containment" of the plasma. The reduction of the thrust chamber efficiency due to wall losses will be discussed for both cases of a gas without magnetic fields (*i.e.*, ordinary containment), and a gas contained magnetically.

Ordinary Boundary Layer: In a conventional chemical rocket, losses to the walls are only a small fraction of the total energy flux because of the high Reynolds number of the flows. The high Reynolds numbers are associated with high gas densities and correspondingly high power levels. Electrical propulsion devices on the other hand operate at much lower power levels which result in lower gas densities and lower Reynolds number. The boundary layers on the walls are therefore thicker and a larger fraction of the flow energy is lost to the walls.

For a given shape nozzle the ratio of the boundary layer thickness to the nozzle diameter can be shown to vary as

$$\frac{\text{Boundary layer thickness}}{\text{nozzle diameter}} = \frac{\text{characteristic length}^{1/2}}{\text{total thrust power}} (\text{Specific Impulse})^{3/2}.$$

Since the gas in the boundary layer has transferred its energy to the walls, the ratio of boundary layer thickness to nozzle diameter is roughly the fraction of the energy which has been lost to the walls. From the above equation it is clear that in order to reduce these losses the characteristic length of the nozzle should be small and the power level should be large. As one goes to higher specific impulses, the power level must be increased or the characteristic length reduced in order to maintain efficient operation.

The heat conduction coefficient of a gas in the temperature range of interest is appreciably higher when the gas is fully ionized than when the degree of ionization is small. A large heat conduction coefficient implies thick boundary layers and therefore greater wall losses. The power levels required for efficient operation are therefore larger for a fully ionized gas than for a slightly ionized one. If wall losses are to be kept small when there is an ordinary boundary layer it is therefore preferable to use a gas such as helium which is difficult to ionize rather than lithium which is easily ionized.

Magnetohydrodynamic Containment

For a highly conducting plasma it is possible to apply magnetohydrodynamic forces at the boundary of the plasma in such a way that a magnetic bottle is formed which prevents the plasma from coming in contact with the walls. The thickness of the surface layer in which this force acts is closely related to the skin depth in which currents flow when an ac electromagnetic signal is incident upon a solid conductor. A close analogy may also be drawn between the thickness of this region and the thickness of an ordinary aerodynamic boundary layer. Just as an ordinary boundary layer represents the diffusion of heat or vorticity, the growth of the magnetohydrodynamic boundary layer represents the rate of diffusion of the gas through the magnetic field. As in the aerodynamic case, the thickness of the boundary layer represents the losses associated with the containment mechanism. In order to achieve high efficiencies it is therefore necessary to insure that this boundary layer thickness is small compared to the nozzle diameter.

The boundary layer thickness decreases rapidly with increasing temperature (or specific impulse), because of the rapid increase in the electrical conductivity with temperature. Magnetohydrodynamic containment is therefore easier at high specific impulse.

It can be shown that in contrast to ordinary boundary layers, the thickness of the magnetohydrodynamic

boundary layer is independent of the gas density. For a given specific impulse the ratio of the boundary layer thickness to channel size will decrease as the channel size is increased. For efficient magnetohydrodynamic containment it is therefore desirable to make the nozzle as large as possible. The upper limit to the size would probably be determined by the allowable mass of the thrust chamber.

The two basic types of containment which have been considered compliment each other in some respects. For example, losses in the ordinary boundary layers are less at low specific impulses, while magnetohydrodynamic containment is more efficient at high specific impulses. Magnetohydrodynamic containment is easiest for fully ionized gases where the magnetic forces act directly on all the particles, while ordinary boundary layers are most efficient for partially ionized gases which have a lower heat conductivity.

Electrode Losses: For systems in which there are electrodes, the current flowing to the electrode surfaces produces losses. In conventional arcs, drops of about 20 volts exist in the immediate neighborhood of the electrode surface. The power associated with the current flowing to the electrode through this potential drop is dissipated in the electrode and must be considered as a loss. Since these potential drops are in series with the potential across the gas, the ratio of the power loss at the electrode surfaces to the useful power is the ratio of the voltage drop at the electrode to the voltage across the gas. Therefore, in order to attain high efficiencies the voltage drop in the gas must be large compared to the electrode voltage drops.

In the arc jet, the voltage across the gas is given by the product of the current and the resistance of the gas. At low temperatures the conductivity of the gas is low and therefore the resistance high. In this case the voltage drop in the gas column can be made larger than the electrode drops and a high efficiency can be achieved. However, as the arc is pushed toward higher specific impulses the gas temperature and conductivity increase resulting in smaller voltage drop in the arc column. Therefore, at high specific impulses the arc jet may become inefficient due to electrode losses. It can be shown that inefficient operation due to frozen flow losses probably occurs at a lower specific impulse.

Electrode losses can be appreciably reduced in devices using magnetohydrodynamic forces. The motion of the gas in the magnetic field will generate a back emf, similar to the back emf induced in the armature of an electric motor. This back emf is independent of the gas conductivity and is in series with the other voltage drops across the gas. For such devices it is therefore possible to achieve large voltages across the gas even in cases of high electrical conductivity.

The magnitude of the back emf which can be achieved will depend upon the strength of the magnetic field which can be used. This in turn will be limited by the character of the flow in the thrust chamber.

The three major types of losses which have been discussed for neutral plasma thrust devices can limit the specific impulse operating range. In general, the thrust chamber efficiency tends to improve with increased power level. It may therefore be desirable to store electrical energy and use it in pulses in the thrust chamber. In this way the instantaneous power level can be higher than the average power level of the electric power supply.

Plasma Propulsion Possibilities*

WARREN RAYLE†

Summary—**Plasma propulsion systems are arranged in three categories according to the cyclic nature of the jet produced; 1) steady, 2) alternating or wave-accelerated, and 3) pulsed. Typical systems within each category are described with emphasis on work being done at the NASA Lewis Research Center. The criteria for evaluating the relative merits of these systems are listed and the advantages and disadvantages pointed out. Such parameters as energy efficiency and propellant utilization must be measured before conclusions can be reached as to the eventual applicability of any of these systems.**

Introduction

THE achievement of manned, interplanetary space flight depends to a large extent on the development of reaction motors having exhaust velocities in the range of 10^6 to 10^8 centimeters per second. Such velocities can be obtained by the use of electric and/or magnetic fields to accelerate charged particles, ions or plasmas. This paper is directed towards describing some of the plasma acceleration schemes which have been proposed and particularly to the work being carried out at the NASA Lewis Research Center. This type of research was largely initiated during the past year, and much is in its early phases.

A theoretical analysis of the propulsion requirements for spacecraft[1] shows that the optimum jet velocity for a particular mission is strongly dependent on the parameters a_0, the ratio of thrust to initial gross weight, and α, the powerplant specific weight in pounds per jet kilowatt. Generally, a reduction in powerplant weight permits a corresponding increase in acceleration, or a_0. For a typical mission, earth-orbit to Mars-orbit and return, the value of the combined parameter αa_0 must be less than about 0.004. For such a mission accelerations as low as 10^{-3} or 10^{-4} are feasible—the trip time is not greatly increased. For a fixed value of a_0 the optimum jet velocity corresponds to a minimum combined weight of powerplant and propellant. High velocities require a large powerplant, and low velocities a large mass of propellant. Although the optimum velocities are found to be of the order 3×10^6 cm per second and greater, it is possible to accomplish such a mission using nonoptimum velocities as low as 0.8×10^6 cm per second. Velocities up to about 1.5×10^6 cm per second might be obtained by thermal expansion of a heated gas, *e.g.*, a nuclear rocket. Since temperatures of at least 50,000° R are needed to reach 3×10^6 cm per second, it seems necessary to investigate other accelerating systems.

The use of a beam of ions for propulsion is subject to certain limitations in the thrust that can be achieved per unit area. The use of a plasma as a propellant offers the advantage of permitting higher densities and consequently more thrust. An added advantage is the ability of a plasma accelerator to be adapted to many propellant compositions.

Many different schemes for the acceleration of plasmas have been proposed and even demonstrated. In recent years especially, the interest in nuclear fusion has stimulated a great deal of work in the production of extremely high temperature plasmas.[2] However, much of this work has been aimed at obtaining research tools —not propulsion devices. Criteria for evaluating the relative merit of plasma propulsion systems must include the following: 1) total weight of the system per pound of thrust, 2) efficiency of conversion of input electrical energy to jet directed kinetic energy, 3) propellant utilization, 4) reliability of the system for substantially continuous operation over periods up to a year, 5) values of jet velocities obtainable, and, of course, 6) cost. In determining system weight, provision must be included for disposing of heat generated within the system. Energy losses which go to heat the accelerator are much more serious, therefore, than similar losses arising from random or imperfectly directed velocities in the jet. Plasma propulsion systems may be grouped into three categories, depending on the cyclical nature of the jet produced: 1) steady flow, 2) wave accelerated, and 3) pulsed. Research in these categories will be separately discussed.

Steady Flow Systems

A continuous flow of plasma may be obtained by the simple expedient of expanding the hot gases produced by a stabilized electric arc. Such a fluid-stabilized jet has been developed commercially for a number of applications other than propulsion.[3] The hot gases are drawn off through a hollow, cooled electrode and may then be expanded as in a rocket motor. The propellant is supplied to the chamber within which the arc is located in such a manner that the inward flow of propellant produces a "thermal pinch" effect; that is, the cross section of the arc is kept small by the inward radial flow. The energy of the arc is thus concentrated, and higher effective temperatures may be attained. Veloci-

* Manuscript received by the PGMIL, February 5, 1959.

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[1] W. E. Moeckel, "Propulsion Methods in Astronautics," First Internatl. Congress of Aeronautical Sciences, Madrid, Spain; September, 1958.

[2] A. S. Bishop, "Project Sherwood, The United States Program in Controlled Fusion," Addison Wesley Publishing Co., Reading, Mass.; 1958.

[3] G. M. Giannini, "The Plasma Jet and Its Application," Office Sci. Res., TN 57–520; 1957. Also "The plasma jet," *Scientific American*, vol. 196, pp. 80–84, 86, 88; August, 1957.

ties so produced may approach 10^6 cm per second, but it seems unlikely that 10^7 or 10^8 can be reached. The problems of electrode erosion and the relatively low velocities attainable would seem to impose definite limitations on the applicability of such a system.

Another system which does not seem to be subject to the same limitations involves the radio frequency heating of a plasma to very high temperatures, followed again by expansion. The electrodes, not being present, do not erode. The attainable temperatures are not known, though theoretical analysis at this laboratory indicates that the efficiency of energy transfer into the plasma begins to decline when the conductivity approaches that of a metal. Preliminary experiments with a 3-kw, 30-mc power supply are being carried out to explore the potential of this system.

The high velocities required may also be obtained from a rather simple electric arc, where at certain conditions the electrode material is observed to be expelled at velocities higher than the arc temperature would indicate.[4] Such a jet is being investigated in the configuration of the homopolar motor where the arc is caused to rotate in the annulus between concentric electrodes by an impressed magnetic field as shown in Fig. 1. Jet velocities up to about 1.5×10^6 cm per second have been obtained from a stainless steel, water-cooled cathode. The phenomenon is not well understood and consequently the ultimate potential of such a device is not known. The correlation of jet velocity with electrode melting point[4] would indicate some thermal process to be important, yet thermal expansion alone does not seem a sufficient explanation. The jet velocities have been measured directly from streak photographs, and also calculated from the thrust indicated by the pendulum and the mass flow rate determined from the weight of material deposited thereon. The energy efficiency was calculated to be only 3–4 per cent, which cannot be considered encouraging.

The indicated velocities correspond to kinetic temperatures approaching 10^6 °R, assuming singly-ionized iron, or particle energies of about 60 ev. Since the total arc voltage was of this order, there seems a high probability that electrostatic forces played a dominant role in the acceleration. By the proper choice of electrode material, and by proper spacing of the electrodes, the energy efficiency can undoubtedly be increased. However unless efficiencies exceeding 50 per cent can be obtained, the applicability of such a system seems doubtful.

Another possibility in the category of steady systems would be the production of a plasma jet by the use of combined magnetic and electric fields. The drift velocity induced by an electric field at right angles to a magnetic field is in the same direction both for positive and negative charged particles, and is equal to 10^8E (volts/cm)/B (gauss).[5] This process would be applicable particularly to very low density plasmas.

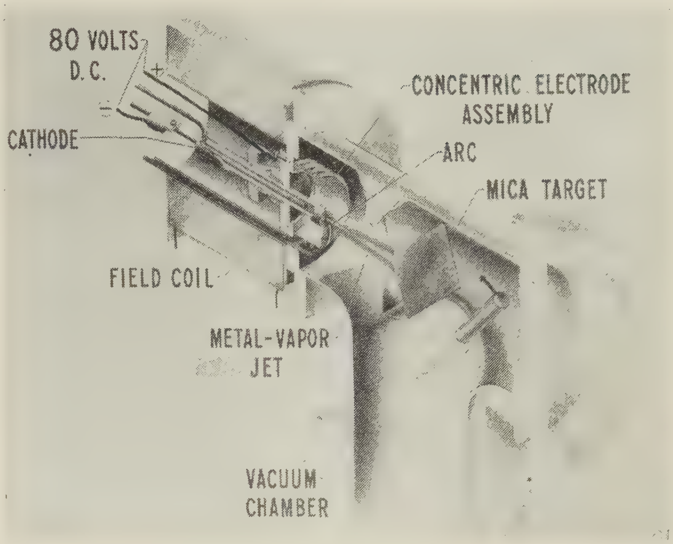

Fig. 1—Apparatus for study of cathode jet from spinning arc.

Other possibilities might include such things as the heating of a very low density plasma at the cyclotron frequency of the ions in a magnetic field, which could contain the plasma during the heating process and also serve as a mirror to convert their velocity to a directed beam.

Wave Accelerators

One device which seems to offer considerable promise as a plasma accelerator might be termed a linear motor, or a moving magnetic mirror. A macroscopic viewpoint could consider that the plasma, being a conductor, appears diamagnetic and hence can be propelled by a time-varying magnetic field. A microscopic view, considering individual charged particles, shows them tending to be reflected from a region of high magnetic field, spiraling along lines of force.[5] The accelerator shown in Fig. 2 would function from either viewpoint. A series of coils energized by multiphase radio frequency current would give the effect of a traveling magnetic wave which accelerates as it progresses due to the increasing coil spacing. A charged particle, or a plasma, would tend to be carried along by this wave. Experiments with a small-scale version of this device are being conducted at this laboratory. About 1 kw at 150 kc is being used with a small number of coils. Field intensities of about 500 gauss within a one-inch tube are produced. These fields are sufficient to form their own plasma by the induced "electrodeless discharge." The plasma material used includes air, helium, and lithium.

Velocity measurements have been made using photo-

[4] E. C. Easton, F. B. Lucas, and F. Creedy, "High velocity streams in the vacuum arc," *Elec. Engrg.*, vol. 53, pp. 1454–1460; November, 1934.

[5] L. Spitzer, "Physics of Fully Ionized Gases," Interscience Publishers Inc., New York, N.Y.; 1956.

cells to determine the phase shift of the luminosity wave along the axis of the duct. Very preliminary data show velocities of the order of 10^6 cm per second obtained with external mean pressures of about 100–1000 microns. The measurements were obscured by a charge separation phenomenon; the time-varying field seems to produce sufficiently high local voltages that a beam of electrons is projected down the tube. Fig. 3 shows such a beam.

The luminosity produced by the electron beam naturally has an apparent velocity of propagation higher than the equipment is capable of measuring.

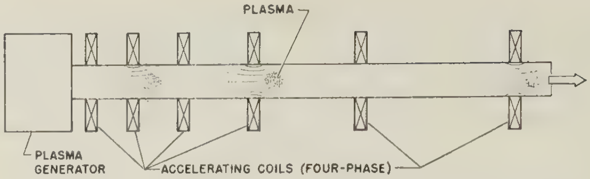

Fig. 2—A traveling-wave plasma accelerator.

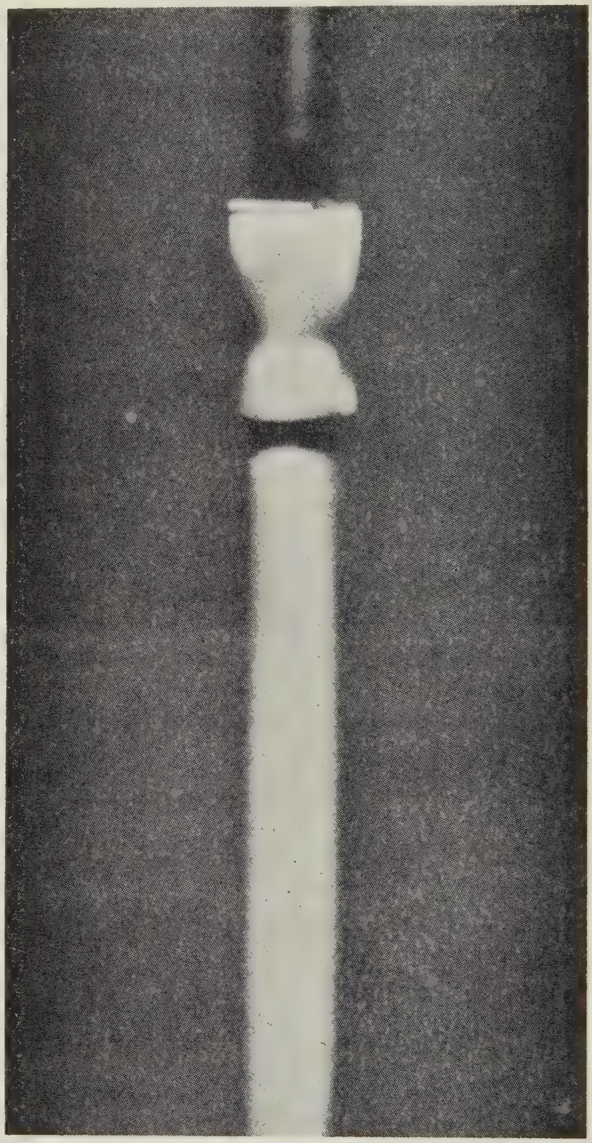

Fig. 3—Discharge produced by a two-coil traveling-wave accelerator.

Pulsed Systems

A number of devices for the production and acceleration of plasma can be most conveniently designed to operate in very brief pulses. Such operation permits the use of currents and accelerating forces much larger in magnitude than would be possible in any continuous or alternating cycle device. Much of this sort of work to date has relied on the condenser-discharge source. One obvious advantage of this type of accelerator is that research can be conducted with a unit which can be "scaled up" for usable thrust merely by increasing the repetition rate of the cycle.

Most of the pulsed systems can be considered as involving the expansion of a current loop, driven by the differential magnetic pressure. One other system, which does not depend on such high currents, is the projection of an arc by means of an external magnetic field.

Currently being studied at this laboratory is a device called the rail accelerator, shown in Fig. 4. An arc is struck between the two rails, at point P. The external magnetic field normal to the arc and to the rails gives rise to the $\bar{j} \times \overline{B}$ force which then accelerates the plasma axially. This device should in theory provide its own periodicity—the ejection of one plasma bunch from the end of the rails permitting the voltage to rise to the level required to strike again at P. The plasma in this case is composed largely of electrode material.

Some of the difficulties are obvious. In the first place, a certain amount of the plasma will be lost to the electrodes during its travel. This not only wastes energy, it also increases the cooling problem. The process of striking an arc in high vacuum is difficult. And it would be inconvenient to supply the propellant in the form of electrodes. By feeding a gas or vapor to the region where the arc is initiated, the arc-striking problem is reduced and the reliance on electrode material for the propellant is obviated. There then appears the problem of insuring maximum utilization of the propellant, since with the system open to space a flow of un-ionized material must be anticipated. One possibility is to inject the propellant in pulses coincident with the arc. Another is to supply the propellant in solid or liquid form—for example, to extrude a filament from one electrode. Since this would involve a much larger mass of material in each plasma

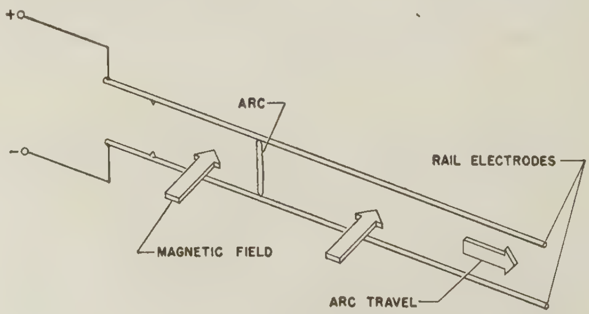

Fig. 4—Rail accelerator.

bunch, higher currents would be required to ionize and accelerate it. With the use of larger currents, the external magnetic field may become small by comparison with the field associated with the current loop. The expansion of a current loop is one system for producing high plasma velocities.

An insight into the possibilities of such a process may be gained by experiments with electrically exploded wires.[6] Such experiments are being conducted at this laboratory using a 2-mfd, 20,000-volt condenser bank. The bank is discharged by means of a thyratron (5563) through a 2–5 mil diameter wire supported between two parallel electrodes similar to those shown in Fig. 4. The ejected plasma impinges on a pendulum. The momentum transferred to the pendulum can be easily calculated from the deflection. If all the impinging plasma condenses on the pendulum, the increase in weight will provide a measure of the plasma mass accelerated. Then the velocity is calculable. Conversely, if the velocity is separately measured the mass may be determined. As yet no direct measurements of the velocities have been made. Very preliminary data indicate that perhaps 10 per cent of the initial mass of the wire is deposited, and the calculated velocity is of the order of 0.5×10^6 cm per second.

Before the feasibility of such a system can be evaluated, information must be obtained as to the parameters governing the extent of ionization of the original material, the energy efficiency obtainable from the system, and, of course, the maximum repetition rate.

Another system for the projection of plasmas at high velocities uses the surface discharge between two closely-spaced electrodes imbedded in an insulating material. Bostick[7] found that such a configuration could produce toroidal plasmas (plasmoids) with velocities of the order 10^7 cm per second. Experiments with a similar device reported[8] by Finkelstein indicate that high velocities may be obtained even with a greater electrode spacing, though the nature of the plasma beam may have been different. Experiments with such a system are being carried out at this laboratory, aimed at determining such parameters as propellant utilization, energy efficiency, etc.

The production of shock waves at velocities of about 10^7 cm per second has been demonstrated by Kolb[9] using a current loop expansion device wherein the initial acceleration forces are enhanced by the close proximity of the plasma to the conductor carrying the current to the arc. Experimental data is again required to evaluate the energy efficiency of such an accelerator, to determine the best method of supplying propellant, and to discern whether a high rate of cycling is feasible.

A system which might be considered a two-dimensional rather than a one-dimensional arc is reported by Janes and Patrick.[10] The arc between concentric electrodes is rotated by an axial magnetic field; the expansion of the current loop (now a surface) is used as a piston to expel a low-density gas from the annulus. Values up to 2×10^7 cm per second are reported. This project is being discussed more comprehensively by Kantrowitz, *et al.*, in a separate article in this volume.

The most obvious difficulty with the whole family of condenser-discharge systems is the condenser itself. While not insuperable, the problems posed by the requirements for light weight, high energy storage, rapid cycling, low loss, and great reliability are certainly serious. Additional problems are involved in the propellant feed system for maximum utilization, and over-all durability. Of course, these systems as well as the others must first of all be evaluated with respect to energy efficiency to see whether this value is at, or can be raised to, a level meriting further development.

Evaluation

At the present time there is no plasma acceleration system which has been shown to be satisfactory for the electrical propulsion or spacecraft. In general, the steady-flow systems not only involve problems of electrode wear, but also are limited to relatively low jet velocities. A possible exception here is the radio frequency heating system, but it is as yet too early to predict the performance obtainable. The wave engines seem good on paper, but may be limited in their efficiency by the I^2R losses in the accelerating coils. The pulse systems which have received perhaps the most publicity are for the most part single-shot affairs, and demand the use of energy storage systems of considerable bulk. Conceivably these may not be intolerable, but this must be demonstrated. Pulse repetition rates desired would fall within the audio frequency range and might well give rise to discomfort on the part of the crew or even to fatigue on the part of the structure.

Research on the heating and acceleration of plasmas is still in its very early phases. More detailed understanding of the behavior of plasmas in various combinations of electric and magnetic fields is needed before a rational approach can be made to the problem of designing a satisfactory plasma accelerator for propulsion. All experimental and theoretical work now being conducted by many organizations on plasma propulsion devices is contributing to this understanding.

[6] W. G. Chace, "A Bibliography of the Electrically Exploded Wire Phenomenon, Advanced Research Laboratory," Geophys. Res. Directorate, AF Cambridge Res. Center, Air Res. and Dev. Command, Cambridge, Mass.; 1956.

[7] W. Bostick, "Experimental study of plasmoids," *Phys. Rev.*, vol. 106; April–June, 1957.

[8] D. Finkelstein, G. A. Sawyer, and T. F. Stratton, "Supersonic motion of vacuum spark plasmas along magnetic fields," *Physics of Fluids*, vol. 1:3, pp. 188–192; May–June, 1958.

[9] A. C. Kolb, "Production of high-energy plasmas by magnetically-driven shock waves," *Phys. Rev.*, vol. 107, pp. 345–350; July 15, 1957.

[10] G. S. Janes and R. M. Patrick, "The Production of High Temperature Gas by Magnetic Acceleration," Avco. Res. Rep. 27; March, 1958.

A Brief Survey of Direct Energy Conversion Devices for Possible Space-Vehicle Application*

A. E. VON DOENHOFF† AND D. A. PREMO‡

Summary—**A brief review is given of various types of devices for converting heat or radiant energy directly into readily available electrical form. These devices include the thermoelectric generator, the photovoltaic cell, the thermionic converter, and the photoemissive converter. The discussion is from the point of view of possible space-vehicle application. An attempt is made to indicate in a general way the present state of development, the advantages and difficulties associated with each device, and to suggest general lines of future research.**

Introduction

TREMENDOUS amounts of energy expended in a relatively short period of time are required to place a space vehicle in orbit. After the vehicle is in orbit, or at least has been given sufficient energy relative to the earth that its path of motion nowhere traverses the denser regions of the atmosphere, a relatively low power source of long duration is necessary if maximum utility of the vehicle is to be realized. For example, one of the more obvious possible uses of an earth satellite is to transmit or relay television and radio signals. Such an application is fully effective only as long as the power source lasts. In order to obtain the most benefit from the huge economic effort required to launch a space vehicle, the life expectancy of the power source should approximate that of the vehicle.

The various energy sources may be classified into three groups: chemical, solar, and nuclear. In most cases, the energy from these sources is available in the form of electromagnetic radiation or heat energy. The form in which the energy is finally desired is usually electrical energy because this form is generally more convenient than any other. The problem then is that of converting heat or radiation into usable electrical energy. The usual process employs a thermodynamic working fluid with a mechanical device to convert a portion of the heat into mechanical energy, and then another mechanical device to convert the mechanical to electrical energy. This process is presently the best understood.

There are, however, devices that transform heat and radiant energy to electrical energy without going through the intermediate mechanical phase, at least on a macroscopic scale. Such devices are known as direct conversion devices. Because of the fact that no moving parts are involved, such devices are essentially simple mechanically and may have an almost indefinite life. Of the various devices, there seem to be four types at the present time that have been developed sufficiently to be considered as possible practical means for effecting the desired conversion. These are the thermoelectric generator, the photovoltaic cell, the thermionic generator, and the photoemissive generator.

Thermoelectric Generators

The physical phenomena that form the basis of operation of the thermoelectric generator have probably been known for the longest time. These phenomena are the Seebeck effect, the Peltier effect, and the Thompson effect. The Seebeck effect, discovered in 1822, consists essentially of the fact that if a closed circuit be made of two conductors of dissimilar material and one junction is maintained at a different temperature than the other, an electric current will flow in the circuit. The related Peltier effect, discovered in 1834, is that heat is produced or absorbed reversibly at the junction of two dissimilar metals depending on the direction of current through the junction. The Thompson effect is that a temperature difference between two points in a homogeneous substance produces a potential difference between the points. All these effects are basic to the operation of the familiar thermocouple.

One might well ask "Why the sudden interest in thermopiles?" The answer to this question lies in the deeper understanding of the phenomena obtained from relatively recent advances in solid-state physics, and the resulting improvements in output and efficiency associated with the use of new materials. A detailed exposition of thermoelectric effects and their engineering applications is given by Ioffe.[1]

A schematic diagram of a thermoelectric converter is shown in Fig. 1. The use of several different substances connected in series both thermally and electrically is typical of present practice.

Although different for different materials, the thermoelectric voltage per degree of temperature difference between the hot and cold junctions of a particular material varies approximately linearly with temperature. Thus, at some particular temperature, this voltage may be maximized by a proper choice of materials. Factors other than the thermoelectric voltage, however, are also of importance. When one junction is maintained at a different temperature than the other, heat is conducted from the hot junction to the cold junction. This conduc-

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[1] A. F. Ioffe (A. Gelbtuch, trans.), "Semiconductor Thermoelements and Thermoelectric Cooling," Infosearch Ltd., London, Eng.; 1957.

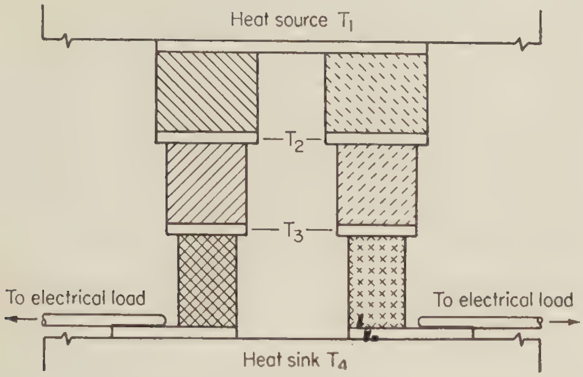


Fig. 1—A possible three-stage thermocouple arrangement.

tion represents an energy loss. Another important source of loss is the Joule heat developed within the material. Detailed analysis of the problem shows that the various effects can be combined into a single parameter that is indicative of the effectiveness of the material in a thermoelectric device. This parameter is

$$Z = \frac{a^2}{\kappa\rho}$$

where

a thermoelectric voltage per degree,
κ thermal conductivity, and
ρ electrical resistivity.

Typical variations of the parameter Z with temperature are indicated in Fig. 2. By choosing proper materials and by adjusting the geometry of the various blocks making up the generator so that each junction is operating near its most effective condition, the efficiency of the thermoelectric generator can be made many times that of the usual thermocouple.

A figure of merit that is frequently used in comparing thermoelectric generators is the ratio of the generator output to that of an idealized machine operating with Carnot cycle efficiency between the same temperatures. Efficiencies of the order of six per cent of the Carnot efficiency have been achieved, and the hope expressed that future research may raise this number to 10 or 12 per cent of the Carnot efficiency. At present, the weight of a thermoelectric generator is approximately 1 lb per watt or 1000 lb per kilowatt. The objectives of research at the present time are to improve this figure by finding suitable thermoelectric material combinations that can work effectively at higher temperatures, thereby improving the ideal efficiency and to find new materials or new methods of using present materials to yield effectively a higher value of the characteristic quantity Z.

Photovoltaic Cells

Photovoltaic cells are of relatively recent origin. They have been used commercially to supply power for some rural telephone lines by converting solar radiation directly into electrical energy. They are also supplying the long-term low-level power on the Vanguard satellite. The general principles of operation of such a cell are illustrated in Fig. 3.[2] Pure crystalline silicon is normally an n-type semiconductor. If enough of an element from Group III in the periodic table, for example, boron, is incorporated in the silicon crystal structure, the substance is transformed to a p-type semiconductor. When the two layers are in contact with no current flowing, the equilibrium condition is for the potential corresponding to the Fermi level in each layer to be the same. The Fermi level is the level that has a probability of one-half of being occupied by an electron. In p-type conductors the Fermi level is close to the top of the valence band, whereas in n-type conductors it is close to the bottom of the conduction band. The boundaries of these bands are, of course, only sharp at a temperature of absolute zero.

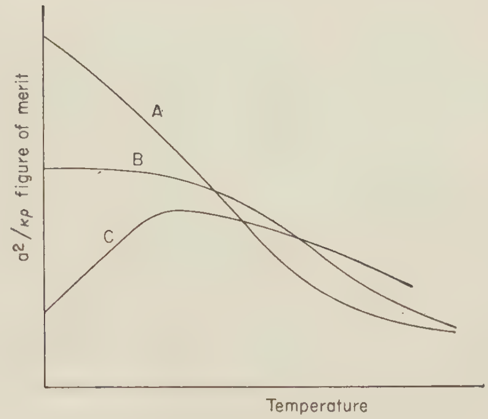


Fig. 2—Typical variation of thermoelectric figure of merit with temperature for various substances.

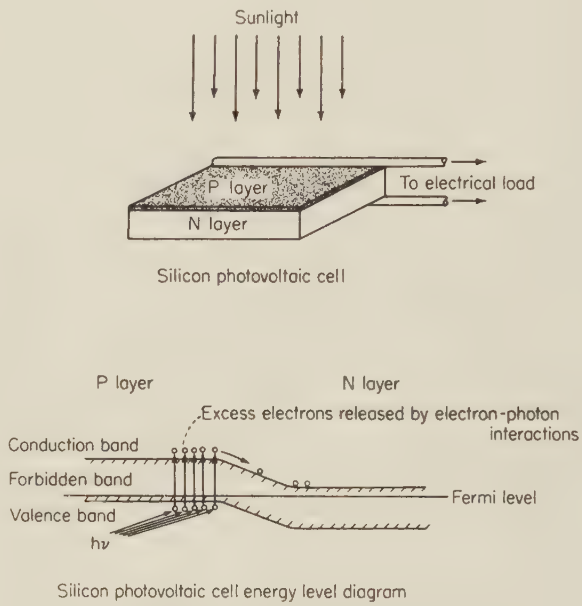


Fig. 3—Schematic and energy level diagram for a silicon photoelectric cell.

[2] P. Rappaport and J. J. Loferski, "New solar converter materials," *Proc. Eleventh Annual Battery Res. and Dev. Conf.*, U. S. Army Signal Engrg. Labs., Ft. Monmouth, N. J., pp. 96–99; May, 1957.

The typical silicon cell consists of a silicon crystal with from one to two square centimeters of surface area into one face of which boron has been diffused to form a *p* layer about one micron thick. The total thickness of the crystal is about one-half millimeter. Photons in the visible range penetrate the crystal to a depth of the order of a micron. When a photon is absorbed, a hole is formed in the valence band and an electron is added to the conduction band. The result is the same regardless of whether the photon is absorbed in the boron-silicon layer or in the pure silicon. In either case, the electron is available to travel through an external circuit connecting the *n* layer to the *p* layer. However, excess electrons in the *p* layer near the *p-n* junction are attracted to the lower energy level of the *n* layer conduction band, thus creating a potential difference between the layers. This potential difference is used to force the electrons through an external circuit. As the temperature is increased, the limits of the various bands become less distinct. This spreading of the band limits results in an increase in the number of electrons in the conduction band, and simultaneously a decrease in the open circuit potential. Thus, the internal resistance of the cell decreases but the generated voltage also decreases with increasing temperature. As compared with an ambient temperature of about 100°F, silicon photovoltaic cells suffer a severe decrease in maximum power output if the cell temperature increases. In fact the maximum power output decreases by about 50 per cent as the cell temperature increases from 100 to 300°F. Performance characteristics of a sample silicon photovoltaic cell are shown in Fig. 4. These characteristics were obtained with the illumination held constant at a value of about one-fourth that of the sun at ground level.

The conversion efficiency of photovoltaic cells can be fairly high, up to about 10 to 12 per cent, if the load impedance is carefully matched to that of the cell at its operating conditions of irradiance and temperature. Under ideal conditions, the weight of the silicon in the cells would be about 20 lb per kw if a cell efficiency of 10 per cent is assumed and the irradiance is 1.4 kw per square meter, approximately the irradiance outside the earth's atmosphere. Unfortunately, the silicon crystal that makes up each cell is extremely fragile. Furthermore, at the present time, the output of each cell is only of the order of 0.01 watt at approximately 0.5 volt. These facts together with the necessity for keeping cell temperature low for higher efficiency, mean that a substantial supporting structure is required in order to obtain a device sufficiently rugged to withstand ground handling and launch, and that the necessary wiring can be extensive and relatively heavy. Estimates have been made of the weight of a silicon cell array sufficiently extensive to have an output of a few kilowatts when exposed to solar radiation outside the earth's atmosphere. If present commercially available cells are used, the unit weight would be about 350 lb per kw. Developmental-type research might reduce this weight to about 200 lb per kw. The use of newly developed more effective materials and the possibility of increasing the size of the individual crystals would, of course, make possible further substantial reductions in unit weight.

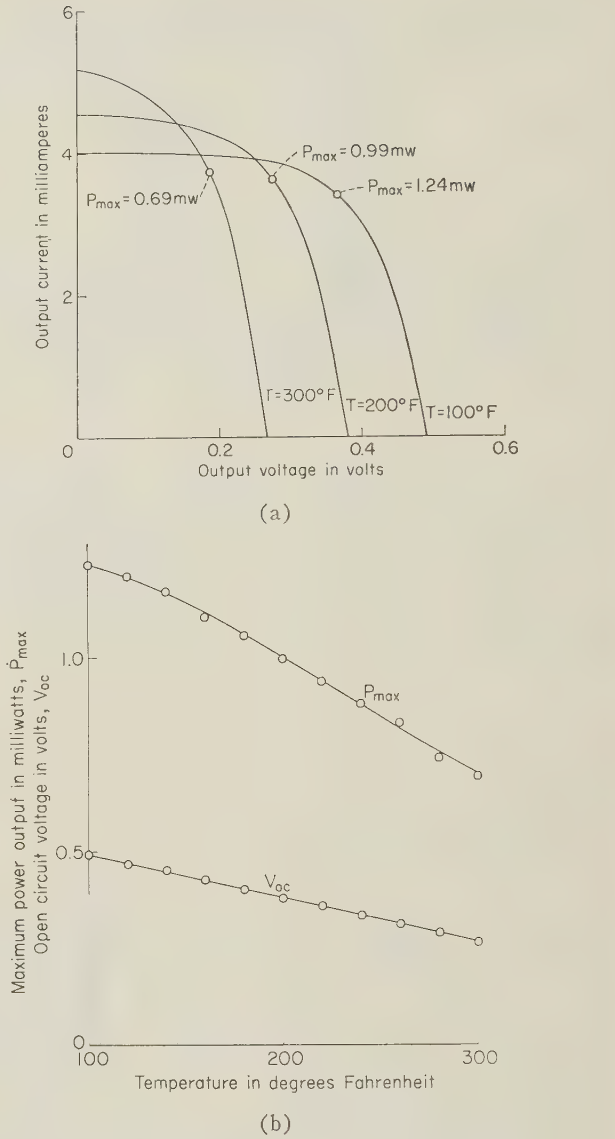


Fig. 4—(a) Typical output characteristics of a silicon photovoltaic cell with constant irradiance; (b) Maximum power output and open circuit voltage vs temperature for a typical silicon photovoltaic cell with constant irradiance.

A little thought will show that if the leads form a substantial part of the total weight, the weight per unit power will increase with increasing power. Since the inverse is also true, photovoltaic cells are comparatively better in small sizes. It is in this range of small powers that, in fact, photovoltaic cells have been used in the past.

Thermionic Converters

Thermionic converters are essentially diodes. The possibility of using a diode as a heat engine was first investigated in detail by Hatsopoulos in his doctoral dis-

sertation.[3] Further development of this subject is given.[4–8]

A simplified sketch of a vacuum thermionic cell and the corresponding energy-level diagram is given in Fig. 5. When the cathode is heated, some of the electrons absorb enough energy to overcome not only the internal cathode work function (ϕ_c) but also the potential field of the previously emitted electrons (space charge). These electrons of relatively high energy then continue to the anode where they lose their kinetic energy plus potential energy of amount ϕ_A electron volts per electron. If the resulting energy level E_{F_A} of the electrons in the anode is more negative than the original level E_{F_c} in the cathode, the difference in energy level, V_0, can be used to drive the electrons through a machine and thus do useful work. Since the diode current is limited by the space-charge potential field, one of the most serious problems connected with this device is the problem of reducing the peak space-charge potential to an optimum level while maintaining a high conversion efficiency. A part of this problem is that of reducing the difference between the energy level at the anode surface and the peak energy level in the space charge region. Several means have been proposed and some have been tested experimentally. An examination of the laws for the distribution of space charge about an electron emitter, which were formulated by Langmuir,[9] indicates that one way of reducing both the peak space-charge potential and the difference in energy level between the peak space-charge level and the level at the surface of the anode is to put the anode very close to the cathode. This method, in fact, has been investigated by Hatsopoulos and Kaye[5] and by Webster and Beggs[8] for spacings as close as 0.0005 inch. The degree of success attained is illustrated by the results presented in Fig. 6, taken from Hatsopoulos and Kaye.[5] The power output is plotted against the output voltage in Fig. 6(a), and the thermal efficiency of the device is plotted against the output voltage in Fig. 6(b). The relatively high power density is to be noted. This characteristic means that the specific weight of such a converter should be

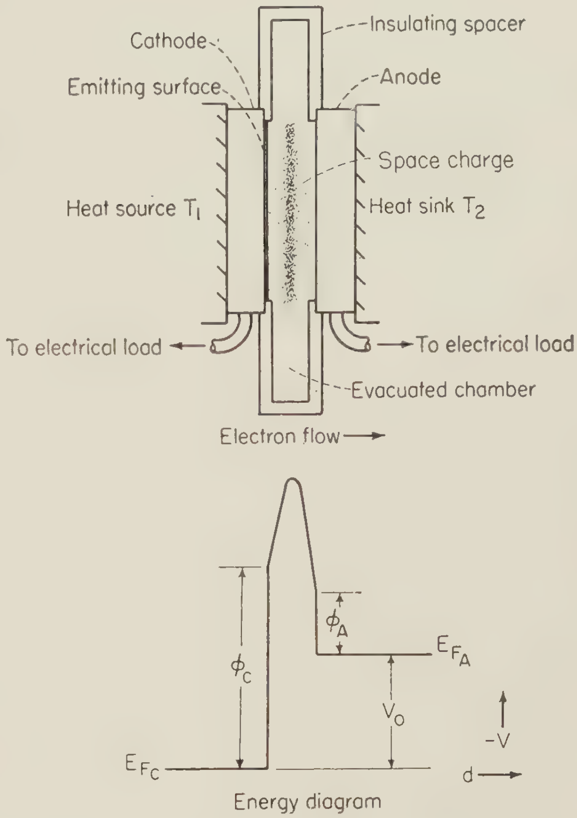


Fig. 5—Vacuum thermionic cell schematic and energy diagram.

relatively low. This is particularly true in view of the fact that no structure is needed to maintain a vacuum in outer space since the ambient pressure at an altitude of a few hundred miles is much less than the lowest vacuum attainable on the ground with presently available equipment.

Another method of overcoming the space-charge problem of the thermionic converter is by the introduction of cesium vapor. The process, illustrated diagrammatically in Fig. 7, operates in approximately the following manner. Cesium vapor is introduced into the evacuated chamber of the device. The vapor tries to coat all of the interior surfaces. The cathode, however, is too hot to permit condensation. If the cathode electron work function, that is, the work necessary to remove an electron from the surface of the cathode is more than the ionization potential of cesium, then when a neutral cesium atom hits the cathode, the atom will bounce back from the hot surface as a positively charged cesium ion. An accumulation of such ions in the space between the cathode and anode will tend to neutralize the electron space charge. The distribution of potential between the cathode and anode is then changed from that shown by the dotted line in Fig. 7 to that shown by the solid line. The advantages of this method are that cesium will collect on the surface of the anode to, in general, give it a lower work function and that the spacing between the anode and cathode can be large compared to the vacuum diode.

[3] G. N. Hatsopoulos, "The Thermo-Electron Engine," Doctor of Science dissertation, Mech. Engrg. Dept., Mass. Inst. Tech., Cambridge, Mass.; 1957.

[4] G. N. Hatsopoulos, "Thermodynamics of Electron Engines," (in press).

[5] G. N. Hatsopoulos and J. Kaye, "Analysis and experimental results of a diode configuration of a novel thermoelectron engine," Proc. IRE, vol. 46, pp. 1574–1579; September, 1958.

[6] V. C. Wilson, "Conversion of Heat to Electricity by Thermionic Emission," presented at the Amer. Phys. Soc. Summer Meeting, Ithaca, N. Y.; June, 1958.

[7] J. M. Houston, "Theoretical Efficiency of the Thermionic Energy Converter," presented at the Amer. Phys. Soc. Summer Meeting, Ithaca, N. Y.; June, 1958.

[8] H. F. Webster and J. E. Beggs, "High Vacuum Thermionic Energy Converter" presented at the Amer. Phys. Soc. Summer Meeting, Ithaca, N. Y.; June, 1958.

[9] I. Langmuir, "The effect of space charge and initial velocities on the potential distribution and thermionic current between parallel plane electrodes," *Phys. Rev.*, vol. 21, pp. 419–435; April, 1923.

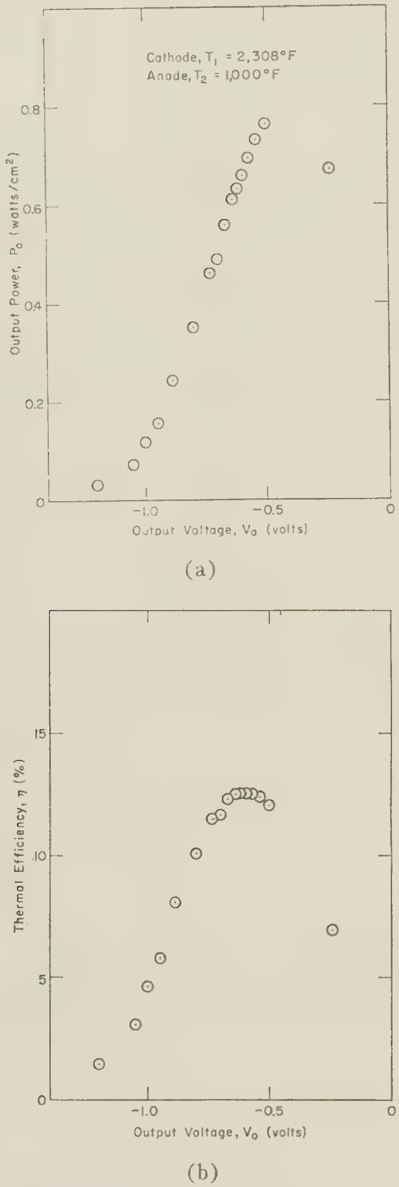


Fig. 6—(a) Experimental power output vs output voltage for diode configuration; (b) Experimental results for sample diode configuration of thermoelectron engine.

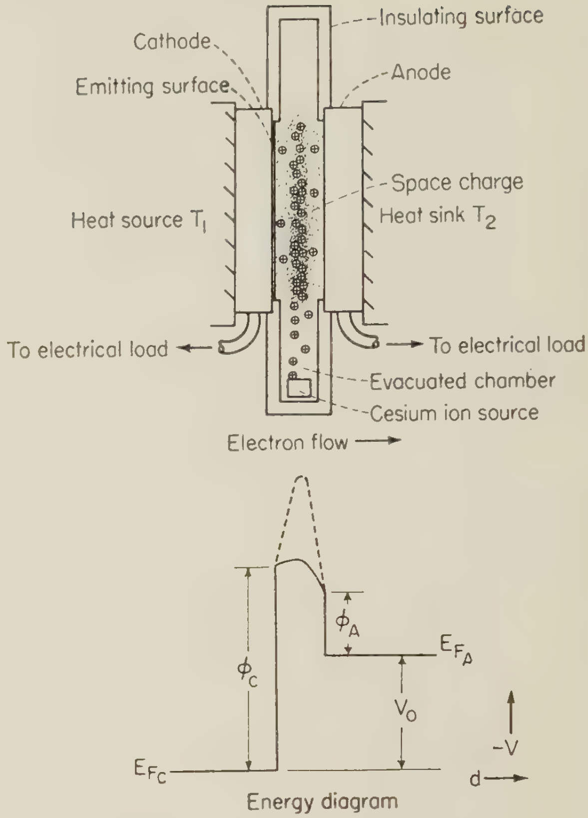


Fig. 7—Thermionic cell space-charge neutralization by cesium ions.

Experimental data on such a cesium vapor diode have been presented by Wilson.[6] In general, the maximum current and power densities were about four or five times those of the high vacuum thermionic converter. For space applications, however, the cesium vapor converter may have certain disadvantages. The converter element, or at least the space between the cathode and anode must be kept gastight. From the point of view of providing a container strong enough to hold the cesium vapor, the problem is not too difficult and would not involve a large weight penalty if it were not for the fact that the wall thickness of the container is dictated not so much by strength considerations as by the condition that it must be able to withstand meteorite impact; as indicated,[10] meteorites can have remarkable penetrating power. The meteorite problem would have to be carefully considered in the design of a cesium vapor thermionic converter for use in a space vehicle.

Two other possible schemes for controlling or neutralizing the space charge have been mentioned in the literature[3] but as yet have not been thoroughly tested. These are: a) the use of crossed electromagnetic fields, and b) the use of a grid as a third element in the device. There is little doubt that converters can be made to operate on either of these principles. The important questions to be answered, and they can only be answered experimentally, are for case a), Is the power required to maintain the electric and magnetic fields so large as to make this method unattractive?; and for case b), Does the current drain to the grid represent too large an energy loss to be tolerated?

The temperatures at which the thermionic converter operate are of interest. For the case cited in Figs. 6(a) and 6(b), the cathode temperature is about 2300°F and the anode temperature 1000°F. The cathode temperature is not so high as to be difficult to obtain or contain, and the anode temperature is sufficiently high that no special difficulty should be encountered in disposing of the waste heat by radiation, the only method available in space. Estimates have been made of the weight of a thermionic converter used in connection with a solar energy collector at from 40 to 80 lb per kw including the weight of the collector.

Up to the present time, the thermionic converters

[10] F. L. Whipple, "The meteoritic risk to space vehicles," in "Vistas in Astronautics," Alperin, Stern, and Wooster, eds., Pergamon Press, New York, N. Y., pp. 115–124; 1958.

have only been built in laboratory-sized units having cathode areas varying from a small fraction of a square centimeter to about one square centimeter. Developmental research is needed to increase the size of the units so that packaged converters delivering power up to several kilowatts are available. Since the efficiency of the converter is greatly affected by the work function of the anode, basic research is needed in an attempt to find substances or combinations of substances having a low work function at the operating conditions of the anode. Much work has, of course, been done on finding suitable substances for cathodes at their normal operating temperatures in connection with vacuum tube filament research. The answers desired in the present case are of a similar nature but for different conditions.

Photoemissive Converter

The last class of converters to be discussed in this paper is the photoemissive converter. This device makes use of the photoelectric effect, that is, the ejection of an electron from a surface upon absorption of a photon of sufficient energy. A schematic sketch and electric potential diagram for a photoemissive converter are given in Fig. 8. Light passes through a transparent conductive coating and is absorbed by the photosensitive substance which forms the cathode. Electrons are emitted from the cathode and collected by the anode. Except for the agent causing the emission of electrons, the method of operation is almost identical with that of the thermionic converter. By the same token, one of the principal problems associated with the operation of the device, namely, control of the space charge, is almost the same as for the thermionic converter. Important practical differences, however, result from the fact that the cathode temperature need not be high. In fact, the cathode temperatures may be sufficiently low that lightweight materials such as plastics can be used for the structural components of the device. Although no experimental data on a practical device have been obtained, the inherent characteristics of the device make it appear very attractive and worthy of detailed investigation.

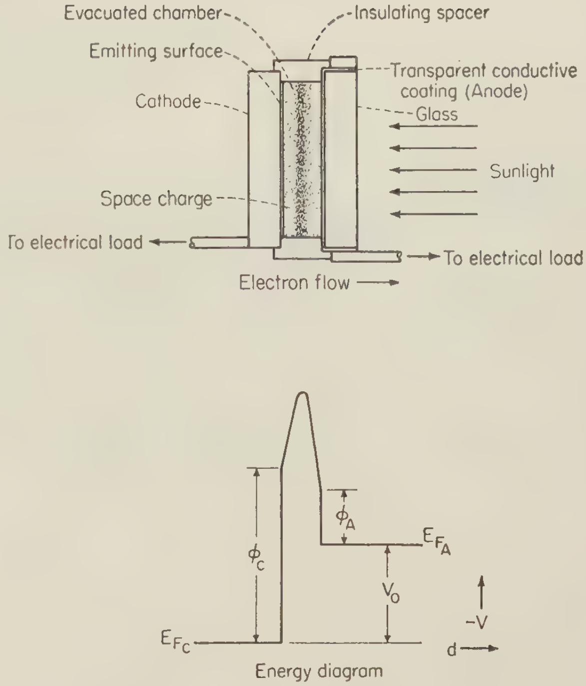


Fig. 8—A possible photoemissive cell.

Conclusion

This brief review has attempted to indicate the types of devices that are presently being considered for use on space vehicles for converting heat or radiant energy to electrical form, and to give some idea of their present state of development. One thought that has not been mentioned is the possibility of combining two or more types of devices into a single unit. This possibility appears to be advantageous in the case of the thermionic converter and the thermoelectric converter. The thermionic converter rejects heat at temperatures of the order of 1000°F. This heat could represent the input to a thermoelectric converter that would later radiate the heat away at a lower temperature, say, 300°F to 500°F, and contribute substantially to the power output of the entire unit.

Fusion for Space Propulsion*

STEPHEN H. MASLEN†

Summary—**The possible role of a controlled thermonuclear reactor in space missions is discussed. Although such a reactor is many years from reality, some of its properties are understood well enough to indicate problems which will appear and which are peculiar to space flight. It appears that it will have to deliver electric power or thrust at a weight of about one pound per kw in order to represent significant improvement over other systems, notably the fission-electric one. One attractive feature of a fusion reactor, as now envisioned, is that it may lend itself to the direct production of electricity or even thrust, without an intermediate heat cycle. It is essential to avoid such a cycle if the weight is to be kept low.**

Introduction

THE biggest problem in the application of a fusion reactor to space propulsion is that there is, as yet, no such reactor. However, there is one basic feature of this ghost which makes it very attractive for a space application. When such a reactor is built, it should produce power at almost no cost in fuel. While two other kinds of power, namely solar and fission, also have this feature, the hope is that the thermonuclear approach may lead to a lighter system. In the present paper, two questions will be examined. First, if there were such a device, what could we use it for? Second, in the design of a thermonuclear powerplant, what problems arise that are peculiar to the space applications?

Missions

The present space propulsion systems, or at least those most nearly advanced to useful form, are the chemical rockets. These are characterized by specific impulses in the range of 200–300 seconds with the possibility of increasing this number to something over 400 seconds. In such a system, the overwhelming part of the weight is the fuel. While there is little doubt that a vehicle powered in this way could accomplish, say, a round trip to Mars, the weight required is very great relative to the payload. Consider, for example, a 4-stage chemical rocket with specific impulse of 420 seconds. Starting in an orbit about the Earth, going to Mars and returning to an Earth orbit along a minimum energy path would require that about 90 per cent of the initial weight be fuel.[1] Remember that in this number there is no allowance for getting the ship into orbit in the first place. Furthermore, for a trip to any other planet, the situation would be even more severe. In the paper just referred to, Moeckel points out that the initial weight in orbit could be cut by a factor of 3 to 5 by using a nuclear rocket or a fission-electric system. Then the useful load might be about half the weight in orbit.

The fission-electric system has been envisioned by most authors[1–3] as one in which a fission reactor heats a working fluid which drives a turbine. This, in turn, drives an electric generator. The power so produced is then used in an essentially separate part of the system to accelerate some propellant to very high velocity, say 400,000 feet per second, which corresponds to a specific impulse of 12,000 seconds. The big advantage of this approach is that the power is generated at almost no cost in fuel, while the propellant is minimized by using a very high exhaust velocity.

Such a propulsion system—one in which energy is generated separately from its use in producing thrust—cannot, in all likelihood, be made light enough so that it could produce a thrust as large as the vehicle weight. That is, it could not take off from the surface of a planet. In order to do so, the ratio of powerplant weight to power produced would have to be less than about 0.01 lb/kw (jet), if the propellant exhaust velocity is to be kept high. If this velocity is not high, the vehicle gross weight will rise sharply and be mostly fuel. The figure of 0.01 lb/kw is smaller than values estimated on the basis of current technology by a factor of almost 1000.

It we cannot take off from a planetary surface, the remaining mission is that of transfer between orbits, say, from an orbit around the Earth to one about Mars. Other uses of such low thrust propulsion systems are to sustain a low altitude satellite or to correct satellite orbits, but the transfer case is the most important. Suppose we consider, to be definite, a round trip to Mars using a vehicle which has a payload of about half the initial gross weight. For optimum performance, the specific impulse should be high enough so that the propellant and powerplant weights are approximately equal. Hence, we limit the powerplant weight to about one-quarter the initial gross weight. The ratio of powerplant to initial gross weight, W_P/W_0, can be written as

$$\frac{W_P}{W_0} = \frac{\alpha P_j}{W_0} = \frac{\alpha F I}{W_0 g} = \frac{\alpha a_0 I}{g}$$

where α is the weight of powerplant per unit of jet power, P_j is the jet power; F, the thrust; I, the specific impulse; g, gravity; and a_0 is F/W_0, the initial acceleration in g's. If α is in pounds per jet kilowatt, and I is in

* Manuscript received by the PGMIL, February 5, 1959.
† Lewis Res. Center, NASA, Cleveland, Ohio.
[1] W. E. Moeckel, "Propulsion Methods in Astronautics," First Internatl. Congress of Aeronautical Sciences, Madrid, Spain; May 12–14, 1958.
[2] M. I. Wilinski and E. C. Orr, "Project Snooper, a program for unmanned interplanetary reconnaissance," *Jet Prop.*, vol. 28, pp. 723–729; November 1958.
[3] E. Stuhlinger, "Electric propulsion for space ships with nuclear power source," *J. Astronautics*, vol. 2, pp. 149–152, 1955; vol. 3, pp. 11–14, 1956; vol. 3, pp. 33–36; 1956.

seconds, this relation gives

$$\alpha a_0 = \frac{45.9 W_P/W_0}{I}.$$

For most cases of interest the optimum impulse is of the order of 10,000 seconds. If, as was suggested earlier, W_P/W_0 is not greater than one-quarter, we have, finally,

$$\alpha a_0 < 10^{-3}. \quad (1)$$

Some representative numbers are shown in Table I. The

TABLE I

a_0	10^{-4} gs	10^{-2}	1	
α	10 lb/kw	0.1	0.001	
Thrust	10 lb	10^3	10^5	For a 50 ton vehicle
Optimum specific impulse	12,000 sec	8,000	4,000	
Jet power	2.6 mw	180	9,000	
Powerplant weight	13 tons	9	9	

last four rows in the table were found with the aid of Moeckel's analysis[1] and apply to a 50-ton vehicle. The optimum specific impulse falls somewhat as the thrust rises and so does the allowable powerplant weight. However, the jet power rises tremendously until, at a thrust level giving an acceleration of one g, the jet power is greater than the power produced by the entire TVA. Weightwise, this requires a very efficient powerplant. The question of whether or not this can be done with a fusion machine will be examined later on. However, let us first examine the other competing systems to see how they fit on this table. The analysis leading to the table doesn't apply to the chemical rocket because of its low impulse and because the energy used is not produced in a manner that leaves us free to control the impulse separately. Suffice it to repeat that the payload could not be the 50 per cent used in Table I, but must be more like 10 per cent.

The fission-electric powerplant described earlier is as light a system as has been seriously proposed for the mission now considered. In sizes in the multimegawatt range, it has been estimated that the weight of powerplant might be about to 5[1] or more[3] pounds per kilowatt. If we assume that at least twenty years of development are available before the fusion reactor is available, it can be guessed that it will have to weigh not more than 2 or 3 lb/kw at a specific impulse of the order of 10,000 seconds to be competitive. Such weights imply maximum accelerations of the order of 10^{-4} g's.

Acceleration of a low thrust vehicle is, of course, very gradual and is time consuming. However, on a Mars trip two factors serve to mitigate this effect. First, the total trip time may be very long, of the order of years in a minimum energy transfer; and, second, there are unavoidable delay times in starting the return trip. One must wait for the two planets to be in the correct relative positions. Some of this time can, without loss, be used in the approach and departure maneuvers at the planet visited.

If the powerplant weight could be reduced to $\frac{1}{2}$ lb/kw or less, while maintaining a specific impulse of 10,000 seconds, this would represent a real breakthrough in space propulsion systems. It would mean [see (1)] that acceleration of the order of 10^{-3} g's would be possible. Such an acceleration would be large enough that trip time would be virtually the same as for high thrust chemical or nuclear rockets. Furthermore, excess energy trajectories would be practical although at somewhat reduced payloads. This could, if sufficient performance improvement were made, lead to greatly reduced round trip times.[4] Of course some steps in this direction can be contemplated even with the presently proposed fission-electric systems. However, it does not seem likely that the fission system can have its weight reduced below a few pounds per jet kilowatt. The heat cycle is too massive; the waste heat rejective system alone is very heavy.

What we would like, then, is a propulsion system which is capable of producing jet velocities of the same order as those obtained with the fission-electric system, but with much less weight. Where does fusion fit in this? It certainly is useless to simply replace the fission reactor in the nuclear-electric system with a thermonuclear one. The weight of the reactor is not great in the first place. Fusion has a place in space propulsion only if it can be used in a manner which does not employ a heat cycle. There are two ways in which this appears feasible.

First, the nature of a thermonuclear reactor is such that it seems reasonable that one could generate electric power directly by expanding the plasma against a confining magnetic field. This might or might not mean a net weight saving, but it certainly would mean that the rotating machinery presently contemplated in the fission-electric case would not be required. This use of moving parts is best avoided if at all possible. Remember that the vehicle must operate for years without a major overhaul, so that long time reliable service is imperative. Furthermore, the absence of moving parts opens the possibility of using higher material temperatures. This is of great importance when, as in space, the waste heat must be radiated and high radiator temperatures are needed to keep the radiator weight down. A second possibility for the fusion reactor is that thrust might be generated directly without the need for electricity.[5] This would presumably be done by expelling the products of the fusion reaction after mixing them with some other auxiliary working fluid.

Fusion Reactors

Now, let us consider the question of whether or not either of these approaches might enable a thermonuclear system to be substantially lighter than the fission one already described. First we should examine in more

[4] W. E. Moeckel, "Interplanetary Trajectories with Excess Energy," Ninth IAF Congress, Amsterdam, Holland; 1958.
[5] M. U. Clauser, "The feasibility of thermonuclear propulsion," in "Proceedings of the Conference on Extremely High Temperatures," John Wiley and Sons, Inc., New York, N. Y.; 1958.

detail the nature of the thermonuclear reactor problems and the presently envisaged solutions.

A fusion reactor is just what the name implies. It is a chamber of some kind where a reaction takes place in which nuclei are fused. The reactions of most interest are those of deuterium with deuterium or tritium:[6]

$$D + T \rightarrow \mathrm{He}^4(3.5) + n(14.1)$$

and

$$6D \rightarrow 2\mathrm{He}^4(7.1) + 2p(17.7) + 2n(16.55) + (1.8), \quad (2)$$

where the numbers in parentheses are the energies carried by the products (in mev). The second reaction above is really a set of four reactions, some of which might not go to completion, particularly in a cyclic operation of the machine. The first of these reactions has the advantage that it proceeds at lower temperatures—10 kev (10^8 °K) is satisfactory—while the *D-D* reaction needs a higher temperature. However, the second reaction is attractive because deuterium occurs naturally, in sea water, while the tritium must be manufactured. For a large surface power station the tritium problem is of great importance. However, for space application, this may not be true. Clauser[5] has proposed a fusion rocket which might require only about one pound of tritium for a 50-ton vehicle on a Mars trip.

In any event, to make the fusion reactions proceed rapidly enough, temperatures of at least 100 million degrees must be achieved. It is apparent that a plasma at this temperature cannot be held in a material vessel. The reason is not that the hot gas would vaporize the walls. There isn't enough gas involved. It is rather that the walls would quench the reaction. The only possibility is to contain the plasma in a magnetic bottle of some kind. The problems and progress in the American program to do this have been admirably discussed by Bishop[7] but will be repeated here very briefly. There are four main tacks being pursued. One of these is the stellerator.[8,9] Fig. 1 shows the components of such a machine as currently envisaged. Essentially the plasma is held in a longitudinal confining field. (Only a part of these coils is sketched; they actually go all the way around the tube.) Such a simple arrangement is hopelessly leaky because the particles drift rapidly across the magnetic field in the curved parts of the tube. To solve this, helical stabilizing coils (again only partially shown) are added. Two means of heating, for different temperature ranges, are supplied. Also a "divertor" is included as a means of reducing the impurity level in the machine and to alleviate the problem of heat transfer to the tube

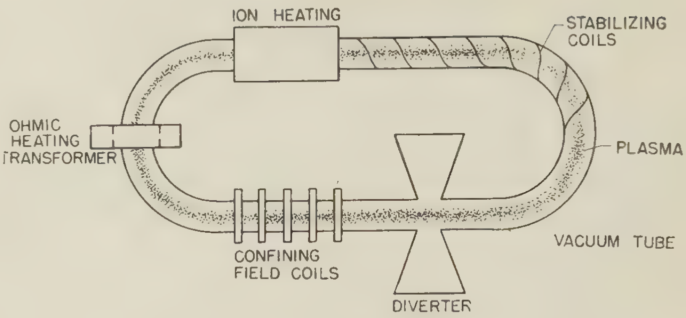


Fig. 1—Stellerator.

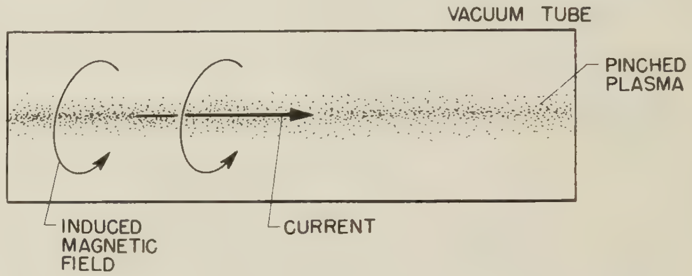


Fig. 2—Pinch.

walls. It is hoped that an operating stellerator will work on a continuous rather than cyclic basis. The approach is promising but appears[6] to lead to very large powerplants, larger than one might need in space. In addition, it seems unlikely that the weight to power ratio can be kept low enough to be interesting for space.

A second approach is to use a pinch[9,10] (Fig. 2). Here a large capacitor is discharged through the gas and the rising current in the gas generates a circumferential magnetic field which pinches the plasma off the walls. To avoid end losses, the tube is often in the form of a toroid. This device, quite obviously, must operate in a cyclic manner. While apparently simple, it suffers grievously from instabilities, although the addition of an external longitudinal magnetic field helps.

The third possible reactor is the pyrotron or magnetic mirror.[9,11] Fig. 3 shows the arrangement which is an axisymmetric one. A straight tube is used and the confining field is generated externally as in the stellerator. In the stellerator, the end problem is solved by making a closed tube. Here the ends are closed by having the field strength several times larger at the tube extremities. These "mirrors" serve to reflect thè charged plasma back toward the center of the tube. This arrangement is most attractive from the point of view of generating electric power. In principle, when the plasma heats and expands against the external magnetic field, electric power can be taken out through the same coils that generated the field. This, of course, implies a cyclic operation.

[6] A. Simon, Nine Lectures on Project Sherwood, ORNL-2285; 1957.

[7] A S. Bishop, "Project Sherwood, The United States Program in Controlled Fusion," Addision Wesley Publishing Co., Reading, Mass.; 1958.

[8] L. Spitzer, "The stellerator concept." *Phys. Fluids*, vol. 1, pp. 253–264; July–Aug. 1958.

[9] Papers presented at the Controlled Thermonuclear Conference, TID-7558, Washington, D. C.; February 3–5, 1958.

[10] W. R. Baker and S. A. Colgate, "A Summary of the Berkeley and Livermore Pinch Programs," presented at Second U.N. Internatl. Conf. on Peaceful Uses of Atomic Energy, Geneva, Switzerland; 1958. (Preprint A/conf. 15/P/1064.)

[11] R. F. Post, "The Pyrotron Program," presented at Eleventh Annual Conf. on Gaseous Electronics, New York, N. Y.; 1958.

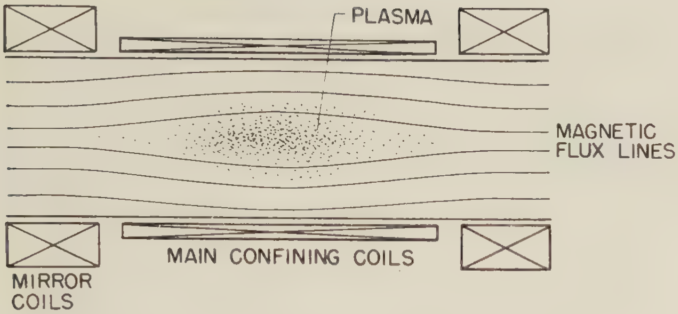


Fig. 3—Pyrotron.

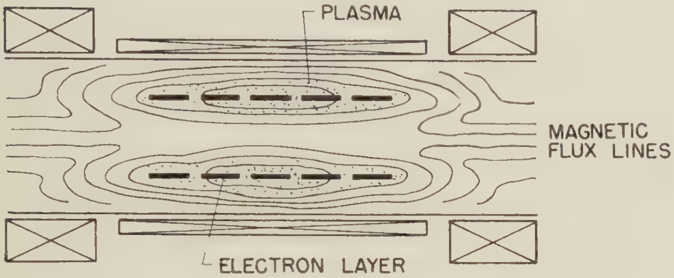


Fig. 4—Astron.

An inherent shortcoming of the pyrotron geometry is that the mirrors leak. The flux lines are not closed and particles whose trajectories are essentially parallel to those lines will not be reflected by the mirrors. A device which overcomes this problem is the Astron.[9,12] Fig. 4 shows the arrangement. The geometry is that of a mirror machine. However, it is proposed to establish a layer of very high energy electrons inside the tube. These relativistic electrons will modify the magnetic field so that a pattern is formed having closed flux lines. Neutral deuterium or tritium is then injected and ionized by the electrons. Steady-state operation is contemplated.

The biggest question with regard to the successful culmination of any of these programs is the problem of stability. All of the devices have many problems, but this is the vital one. A great many stability analyses have been performed, with the result—to vastly oversimplify the conclusions—that under some rather restrictive circumstances stability may be maintained.[13]

Space Problems

If we assume now that the many and serious problems associated with the ground operation of a fusion reactor are solved, we are still faced with the problem of getting it into space. It is clear that the fusion powerplant must have two properties to be worth pursuing. First, we must be prepared to use the system extensively to amortize the development cost, and, second, it must have promise of being appreciably lighter in weight than the alternatives. We will assume that the first condition is met.

Let us consider a fusion machine based on a mirror arrangement, that is, a straight tube with externally powered confining coils (Fig. 3). One can easily show[6] that the power generated in the reacting plasma per unit length is

$$P \simeq 5 \times 10^{-8} \beta^2 R^2 B^4 \text{ kw/cm} \qquad (3)$$

where R is the plasma radius (cm), B the magnetic field strength (kilogauss) at the plasma, and β is the ratio of material pressure to magnetic pressure. The reaction assumed is the D-T one at 100 kev kinetic temperature.

Before going further, it is perhaps worth getting some idea of the size of reactor of interest. Consider a 50-ton vehicle on a round trip to Mars, powered by a low thrust system using a thrust to initial gross weight ratio of 10^{-4}. From Table I, a power output of the order of 3 megw is needed. From (3), we find that a reaction zone two feet long and two feet in diameter is called for if the field strength is 50 kilogauss and $\beta = \frac{1}{2}$. If the field strength were 160 kg, then the characteristic dimension would be about five inches.

The most important weight in this mirror machine is likely to be in the confining coils. If they are made of copper with half the space used, the weight per unit length is

$$W_c = 42(R_1^2 - R^2) \qquad \text{gm/cm}$$

where R_1 is the outer diameter of the coil. If we assume that some fraction α of the power generated in the plasma is available for use, then the weight to power ratio of the coils alone is

$$W_c/P \approx \frac{5 \times 10^5}{\alpha \beta^2 B^4} \left[\left(\frac{R_1}{R} \right)^2 - 1 \right] \qquad \text{lb/kw.}$$

For example, suppose that $\beta = \frac{1}{2}$, $R_1/R = 2$, that 25 per cent of the plasma power is useful and that the field strength is 50 kg. Then

$$W_c/P \approx 10 \qquad \text{lb/kw.}$$

This number is already larger than that estimated for a complete fission-electric propulsion system. However, it is apparent from the equations above that there is a very large return to be had if the field strength can be raised. For example, if the field strength could be made 160 kilogauss with the coil size unchanged then

$$W_c/P \approx 1/10 \qquad \text{lb/kw.}$$

The weight of this machine is, of course, not entirely in the coils, but for a device like the pyrotron, the major part of it may be there unless the field strength can be greatly increased. The magnetic pressure goes as the square of the field strength and, for high fields, leads to large hoop stresses in the coils as well as danger of crushing them. Hence, the design of force-free arrangements becomes of great importance. The question of large loads on the coils takes on increased importance

[12] N. C. Christofilos, "Astron Thermonuclear Reactor," presented at Second U. N. Internatl. Conference on Peaceful Uses of Atomic Energy, Geneva, Switzerland; 1958. (Preprint A/conf. 15/P/2446.)
[13] M. Rosenbluth, "Recent Theoretical Developments in Plasma Stability," Fluid Dynamics Division Meeting, American Physical Society, San Diego, Calif.; November, 1958.

when it is remembered that for most proposals, the loading will be periodic. At the same time, the current carried in the coils will also be very large, thus leading to a serious cooling problem. By cooling the coils to cryogenic temperatures, the conductivity can be increased manyfold, thus reducing the joule heating and hence the cooling load. However, such potential gains must be balanced against the deteriorating mechanical properties of the coil material as well as the more severe heat rejection problem.

This heat rejection problem appears to be a serious one. The only way of getting rid of the heat is by radiation. In the case of the large fission-electric systems as analyzed to date, the radiator is the heaviest single component. Of course, this weight is a strong function of the radiating temperature.

Another area in which a large amount of mass can appear lies in energy storage equipment. If a transient machine is considered, energy must be stored somewhere from one cycle to the next. Most current proposals for fusion machines envisage cyclic operation. Thus, each discharge of power creates energy, some of which is used to start the next. This energy is used either to activate magnetic confining coils (stellator, pyrotron, or astron) or to provide a current discharge in the plasma (pinch). Of course there are other places where electric power will be needed but these two are critical. This means that some sort of capacitor arrangement is needed. Presently available capacitors are too heavy and a drastic improvement is required. Some work has been done with barium-titanate condensers which may aid in the solution of this difficulty.

In a continuously operating machine, the energy storage problem arises in another manner. Stored electricity is needed only for starting. However it seems clear that on an interplanetary mission, some means of restarting the reactor, in the event of a forced shutdown, must be provided.

In a fusion reaction, part of the energy is in charged particles and the remainder is in neutrons (2). For the *D-D* reaction about 40 per cent is in neutrons; for the *D-T* reaction, about 80 per cent. This neutral particle emission raises a serious problem. The energy therein can be used only by capturing the neutrons and using the heat liberated. This leads to the same arrangement (a heat cycle) as that used in a fission reactor and leaves no reason for considering the use of fusion. Hence, we must get rid of the neutrons. Actually, the neutrons can have two troublesome effects. They heat the structure in which they are absorbed and they cause radiation damage. It would be best, of course, if most of them escape the vehicle entirely. This can be arranged if most of the structure lies in a small solid angle as viewed from the reactor. Such structure as is nearby must be light. Large confining field coils would not qualify as light and these would be heated by neutron absorption. Another consideration in this respect is that there is more likelihood of radiation damage to the coils if they are cold rather than warm. This argues against the use of cryogenic cooling to maintain a high electrical conductivity in the coils.

In contrast to fission reactors, there is no gamma emission directly from the reactor. Some will arise in the absorption of neutral particles in the structure, but this will be of low energy and should not raise any serious problem.

If the fusion reactor is to be used as a source of electric power, then it is apparent that some arrangement, involving large coils, for absorbing this power is required. Unless the magnetic field strength can be raised to very large values without increasing the weight of coil and cooling much, the weight of the machine cannot be reduced by anything like an order of magnitude as compared with the fission-electric system. It is just this order of magnitude—or, for that matter, several orders of magnitude—reduction in weight that we really want. As was pointed out earlier, a weight reduction of one order of magnitude would made possible trip times equivalent to those for high thrust chemical rockets, while three of four orders of magnitude reduction are called for to get thrusts equal to the gross weights. This cannot be done with coils so we must consider the possibility of a thermonuclear rocket—one which produces thrust directly.

Such a more or less ideal fusion motor has been envisioned by Clauser.[5] In this case a pinch arrangement is proposed. The plasma is contained in a cylinder or toroid. On reaction, some of the products will diffuse through the confining field and can be mixed with a light propellant, say lithium, which also cools the walls, and the products can then be expanded through a magnetic nozzle to generate thrust. Most of the reaction energy in the charged particles goes into the exhaust gas. A little must be converted into electric energy by means of a coil around the reactor. This is used to pump the propellant and maintain the magnetic throat, but is primarily used to initiate and maintain the pinch. Simon[6] has shown that, speaking roughly, the ratio of energy input to charged particle energy output for the deuterium-tritium reaction at a temperature of 10 kw is

$$\frac{\text{Input}}{\text{Output}} \sim \frac{10^5}{3r^2B^2}$$

where B is the induced confining field in kilogauss and r is the radius of the pinched plasma in centimeters. Thus, for $B=150$ kg and $R=12$ cm, the output is two orders of magnitude larger than the input. In this case it should be observed that because of the way the magnetic field is generated, it is higher at the plasma than at the material walls beyond. This is in contrast to the case of a field induced by external coils, where the field strength decreases toward the plasma.

In the *D-T* reaction, about 80 per cent of the energy is absorbed by neutrons. These must be caught in a shield and the resulting heat disposed of unless the

structure is such that they escape the vehicle entirely. Here the pinch machine has a great advantage over the mirror devices. Except for the need to generate a relatively small amount of electric power, no external coils are needed. In such a case, the structure near the reactor can, in principle, be made light enough so that few of the neutrons are absorbed.

The only electric energy required, except that for auxiliary purposes, is a small amount for generating the next pinch or for starting purposes. Clauser[5] estimates this at something less than one-tenth per cent of the power available. Thus the energy storage is small, there are no large confining coils, the neutrons escape readily and the cooling load is small. If this machine can be made to work, then the weight should be low enough to obtain high thrust with high specific impulse. However, it should be reiterated that it is essential that the thrust be produced directly and that no large coils be introduced. This latter requirement makes a self-induced confining magnetic field—that is, a pinch configuration—mandatory. Whether or not this can be done remains to be seen.

Concluding Remarks

The fusion powered propulsion system discussed herein is one of several novel schemes which have been proposed to overcome the limitations of chemical rockets for space travel. Unless an extremely light weight can be attained, the machine will have to be used in a low thrust device. This means that some other device, such as a chemical rocket must be used as a booster to get the vehicle into orbit and for transfers between the vehicle and a planetary surface. This is a disadvantage of the low-thrust devices because it means that two distinct systems, one for orbital transfers and one for escape from the surface of the planet, must be developed. However this use of distinct systems brings up an attractive feature. In a low-thrust, high-impulse vehicle, a large fraction of the weight is in the powerplant as contrasted with the propellant. This means that when a vehicle returns from an interplanetary voyage, a minimum of weight—propellant—must be brought from the earth's surface to the ship to prepare it for another trip. In the case of a chemical rocket, most of the weight is propellant and this must be replaced for each voyage.

One of the biggest questions about the fusion power supply for space is "When?" It is clear from the literature of the Sherwood Project (the American program for controlled fusion research) that we cannot expect to have an operating reactor in the next few years. However, no fundamental reasons have been uncovered which indicate that we will never have one. It may take 20 years, or 50 years, or even longer, but it is clear that the rewards are great enough that the effort will be made. Although the object of the Sherwood program is not space propulsion, it has been suggested that such application may well be the first for a thermonuclear machine.

In summary then, it is apparent that the thermonuclear reactor can have a very important place in the propulsion of large space vehicles. However, we are a long way from an operating reactor, even on the ground. The main consideration in applying such a reactor to space is that the weight must be kept low and, to do this, we must use the reactor without a heat cycle. This implies either direct generation of thrust or, at least, direct generation of electric power. If the thrust is to be high—of the order of the vehicle weight—the ratio of powerplant weight to jet power must not be greater than about 0.002 lb per kw. For a low thrust system larger weights, up to about 2 or 3 lbs per kw, can be tolerated. The main weight will lie in the confining field coils, if any, in the energy storage facilities, and in the heat rejection equipment if such is required.

Astronautics and Propulsion*

KRAFFT A. EHRICKE†

THE sudden onrush of the space age has drawn considerable attention to concepts of exotic propulsion systems, many of which are, in fact, quite old, as far as the concepts proper are concerned. Ion and photo-electron emission propulsion systems were considered as early as the twenties of this century.[1]

Propulsion is perhaps the most important single factor which will dictate our rate of progress in space exploration and space technology, because it is the art of concentrating energy and converting energy from one form in which it can be stored into a useful form in which it produces thrust. The more effectively energy can be stored and the more efficiently it can be converted, the more payload can be carried on a given astronautical mission and the farther can astronautical missions be extended with a given payload.

The momentum law specifies that the force F acting on a body of mass m is equal to the change with time in the momentum $M = \mathrm{mv}$ of the body, where v is the velocity of body m. Thus the thrust force F must be equal to

$$F = \frac{dM}{dt} = \frac{d}{dt}(\mathrm{mv}).$$

In other words, the mass of the rocket must be changed in order to produce a change in velocity under the effect of thrust. The definition of thrust follows from Newton's third law, namely, the product of rocket acceleration dv/dt and instantaneous mass m of the rocket must be equal to the product of mass change per unit time, dm/dt and departure velocity v_e of the mass change dm/dt with respect to the rocket mass m,

$$m\frac{dv}{dt} = \frac{dm}{dt}v_e = F.$$

The thrust is equal to the exhausted mass (mass consumption) per second, $dm/dt = \dot{m}$ and the exhaust velocity v_e. The propulsive energy must therefore eventually appear in the form of kinetic energy of the exhaust gas:

$$E = \frac{\dot{m}}{2}v_e^2.$$

Every thrust-producing (*i.e.* propulsion) system of a vehicle which operates independent of its environment (rocket) therefore will consume part of the rocket's mass (expendable mass). This is generally true for all propulsion systems, including the so-called photon propulsion system which emits light to produce thrust. In order to produce the beam of light, mass has to be converted to radiation (hence, expended), while the thrust by radiation can be visualized as discharge of photons at light velocity ($v_e = c$), where the mass of the photons, according to Einstein's mass-energy equivalence principle, is $m = E/c^2$, with E being the energy of the radiation.

Since rocket propulsion unavoidably involves a loss of rocket mass, it is an obvious aim of propulsion physics to maximize the exhaust velocity v_e, since only in this manner can the undesirable loss of mass be minimized *i.e.*, a certain thrust $F = \dot{m}v_e$ be produced at small $\dot{m}$. A useful measure for the energy of the jet is therefore its exhaust velocity v_e, or its thrust per unit weight $\dot{w} = \dot{m}g$ per unit time, $F/\dot{w} = v_e/g = I_{sp}$ (lb sec/lb). The thrust per unit weight consumed per second is known as the specific impulse I_{sp} of the expendable material. It is directly proportional to the exhaust velocity, the proportionality factor being the reciprocal of the mass-weight conversion factor, $1/g$.

Thus, increase of the specific impulse means simply increasing the kinetic energy of a given exhaust mass $\dot{m}$. The correlation between specific impulse and energy is

$$I_{sp} = 294.9\sqrt{\Delta E\ (\mathrm{Kcal/gm})} = 133.6\sqrt{\Delta E\ (\mathrm{Kw\text{-}sec/gm})},$$

which means that in order to obtain a certain specific impulse a certain amount of energy ΔE must be converted from whatever its original form to kinetic energy of the gram of exhaust material no matter whether the exhaust material is a combustion gas, a stream of ions, a plasma jet or a beam of radiation. Fig. 1 shows the correlation between converted energy (in kcal/gm and kw-sec/gm, respectively) and specific impulse. A third curve shows the amount of thrust which is produced per kilowatt converted energy at a given specific impulse.[2] Thus, for example, if a quantity of 100 kw of energy is converted in every gram of exhaust mass, discharged per second, a specific impulse of about 1,450 gm sec/gm is obtained, which, in turn, means that about 31 kw must be converted in the exhaust jet every second, in order to maintain a thrust of 1 lb (453.59 gm). From these two numbers which are read from the graph Fig. 1 one finds immediately that, in order to obtain 1 lb of thrust at

* Manuscript received by the PGMIL, March 5, 1959.
† Convair-Astronautics, San Diego, Calif.
[1] H. Oberth, "Wege zur Raumschiffahrt," R. Oldenbourg, Munich and Berlin, Ger.; 1929.

[2] The pressure thrust caused by the gas pressure at the nozzle exit is neglected here. Only the momentum thrust of the gas is considered. The curves in Fig. 1 represent only the energy converted in the jet. In order to obtain the output of the energy source, these values must be divided by the conversion efficiency of the system.

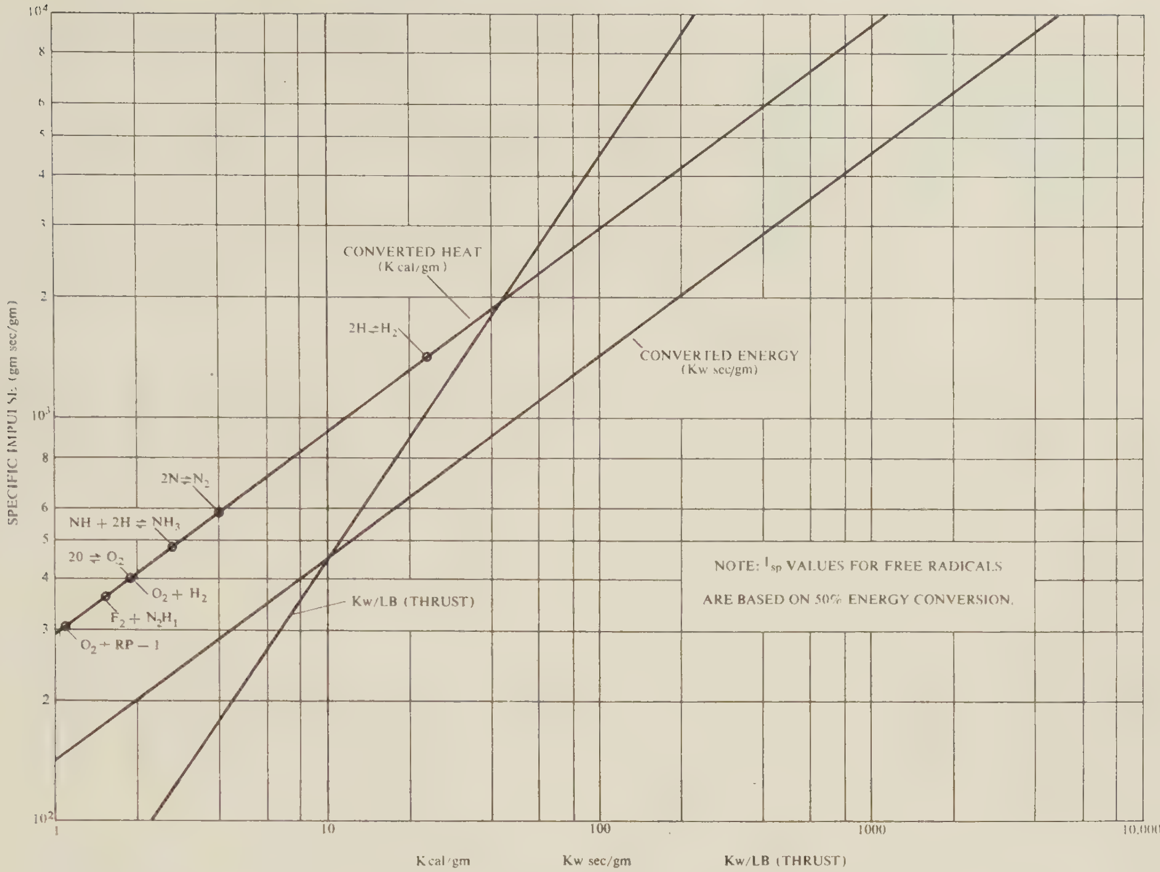


Fig. 1—Energy conversion chart for propulsion systems.

this specific impulse, the exhaust weight must be 31/100 =0.31 gm/sec. Presently, maximum specific impulses of about 300 gm sec/gm are attained in large missiles. Fig. 1 shows that this corresponds to an energy conversion of close to 4 kw-sec/gm or about 7-kw energy converted in the jet per pound of thrust. Thus, a weight of $7/4 \approx 1.8$ gm/sec, or six times as much as before, must be discharged for every pound of thrust. At a 350,000-lb thrust, this corresponds to a loss of weight of 630,000 gm/sec or 1,387 lb/sec of the rocket. This is a terrible toll and forces the rocket engineer to accept a payload weight which is a ridiculously small fraction of the rocket weight, while over 90 per cent of the enormous weight of the present large American and Russian intercontinental and space missiles must be expendable material.

Fig. 1 shows that the specific impulses which can be attained with chemical propellants are small, although the use of such high-energy propellants as oxygen-hydrogen (O_2+H_2) represents a big step forward from oxygen-hydrocarbon (O_2+RP-1; RP=rocket propellant). In view of these limitations of chemical propellants, the interest naturally shifted to other methods of propulsion (*i.e.*, energy conversion). We have three basic energy sources available, namely, *chemical* (from interaction between electrons of electro-positive (fuel) and electro-negative (oxidizer) atoms), *nuclear* (from either fission of the most complex atomic nuclei, or from interaction between the simplest atomic nuclei (fusion)) and *solar* (energy of radiation from the photon source of our solar system). Depending upon the conversion mechanisms, these principal energy sources can be applied in a variety of propulsion systems (Fig. 2). Many of these propulsion systems will be discussed in the subsequent contributions.

Fig. 3 (upper part) surveys the specific impulse ranges of the principal propulsion systems along the $I_{sp}-\Delta E$ correlation explained in Fig. 1. This establishes an arrangement of the propulsion systems in relation to the quantity of energy which they are capable (or believed to be capable) to convert in the jet. The lower part of Fig. 3 arranges the propulsion systems in relation to the thrust-to-weight ratio (*i.e.*, acceleration in Earth-g units) they are known, or estimated, to attain. It will be observed that the acceleration which the propulsion system is capable of giving the vehicle diminishes greatly for specific impulses beyond the range of 1,200–2,000 gm sec/gm or, to put it differently, with the transition from thermodynamic to electrical energy conversion systems (excepting the electric arc system). In fact, their low acceleration restricts their use to interorbital applications (lower right side of Fig. 3). This has a number of reasons. The two principal ones are:

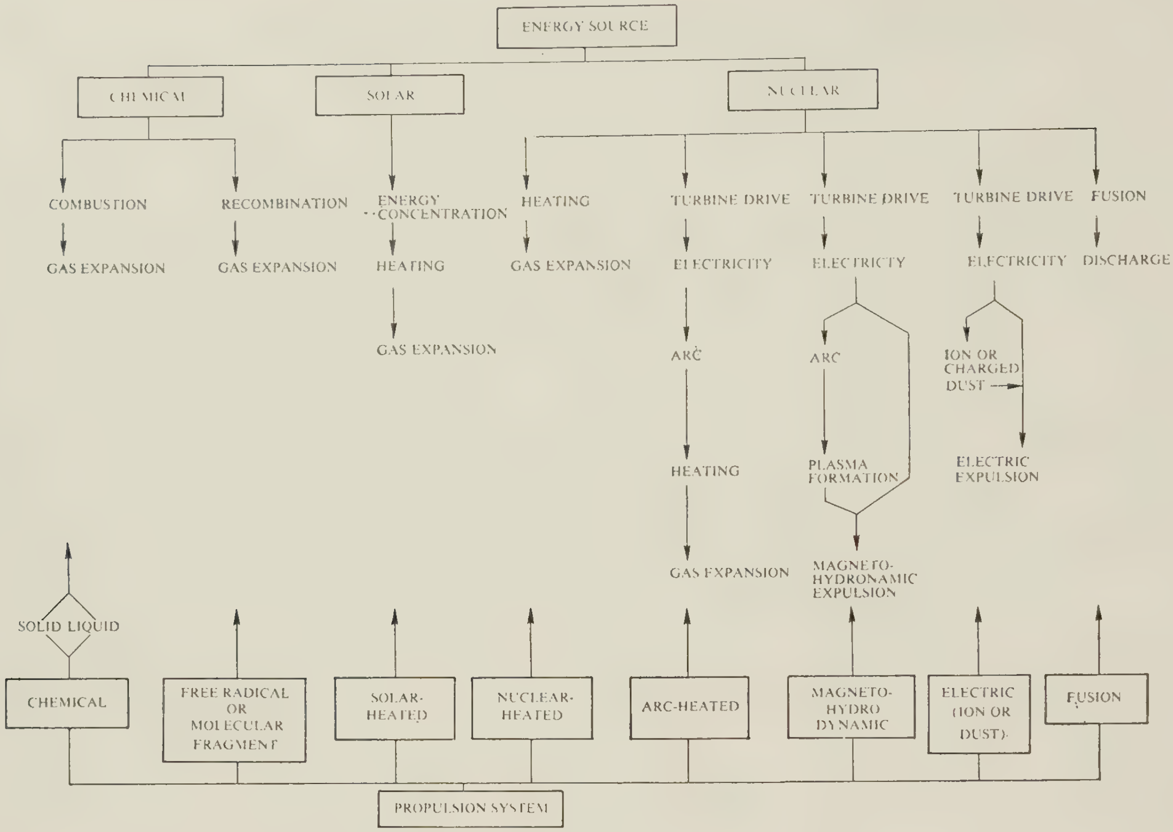


Fig. 2—Application of energy sources to propulsion systems.

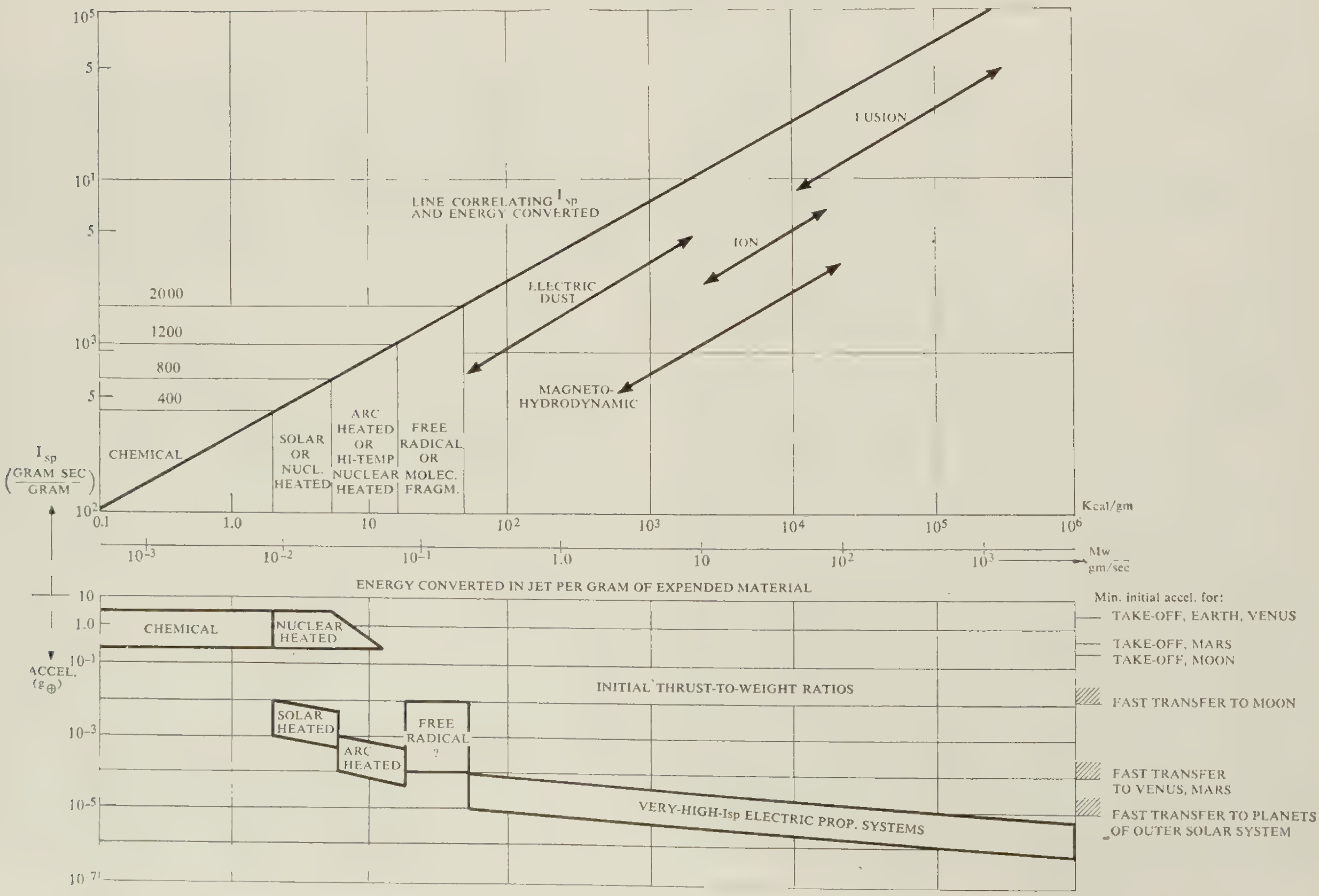


Fig. 3—Propulsion systems and astronautical missions.

1) Increasing quantity of energy which must be converted in the exhaust jet with increasing specific impulse.
2) The electrical systems are characterized by a greater number of conversion steps between primary energy source and exhaust jet (*cf.* Fig. 2). Energy "leaks" out of the propulsion energy conversion system every time a conversion step (*e.g.*, thermal to electrical energy) is inserted, because the efficiency of the process is less than 100 per cent.

This means that the energy output of the primary source must be significantly greater than the energy converted to thrust in the jet (*i.e.*, the over-all energy conversion efficiency of the propulsion system decreases) and that one faces an increasingly difficult problem of getting rid of the excess energy (heat) which leaked out of the propulsion system and into the over-all vehicle system. The first consequence means a bigger and heavier nuclear pile plus auxiliary equipment. The second consequence means an increasingly heavier and clumsier structure, since in space heat can be eliminated from a system only by means of radiation, and this requires surface area. The problems of shutting down a nuclear pile after very high energy output, or of conducting excess heat over very large surfaces fast enough to avoid strongly inhomogeneous heating, hence warping, of the radiation surface and its protection against meteoritic damage, pose additional thorny operational problems. Present solutions tend to increase the weight for a given thrust, hence to reduce the acceleration capability of the propulsion system. Since there are practical limits to the controllable specific (*i.e.*, volumetric) energy output of a fission pile and to the size and effectiveness (heat distribution capability) of radiation cooling surfaces, and since there is a well-defined amount of energy which must be converted per unit weight of the exhaust jet, it follows that these practical limits inevitably limit the weight $\dot{w}$ which can be discharged every second at the given specific impulse. Therewith is defined the limitation of the thrust $F=\dot{w}I_{sp}$. With increasing specific impulse the manageable mass consumption per second becomes smaller and the propulsion systems weight heavier, so that the acceleration decreases with increasing specific impulse. This is the way it looks within the framework of present technology. The most promising way out of the dilemma is to improve the technique of energy conversion, so as to reduce the number of conversion steps in electrical propulsion systems, thereby increasing the over-all efficiency and significantly reducing the weight per unit thrust of the system. In other words, the problem with very high specific impulse propulsion systems is not so much a problem of producing the specific impulses, as a problem of achieving high thrust-to-weight ratios and energy conversion efficiencies (*i.e.*, the ratio of energy converted in exhaust jet to the energy either released by chemical combustion or collected by solar-heating reflectors or produced by the nuclear pile proper). Representative values for the performance and conversion efficiency of some propulsion systems are shown in Table I.

TABLE I

PERFORMANCE AND EFFICIENCY OF SOME PROPULSION SYSTEMS*

Type	I_{sp} (gm sec/gm)	Initial Acceleration (n_0) ($g_\oplus$)	Energy Conversion Efficiency, η_0 ($-$)
Chemical	250– 420	>1.5–0.2 (space)	0.6–0.85 (high expansion)
Nuclear-Heat	600– 1,200	1.5–10^{-2} (space)	0.6–0.85 (high expansion)
† Solar-Heat	700– 800	10^{-4}–10^{-3}	0.6–0.7
† Arc-Heat	1,400– 2,000	$5\cdot10^{-5}$–10^{-4}	$\sim$0.1
† Ionic	$\gtrless 20,000$	10^{-6}–10^{-4}	0.05–0.1
† Plasma (Magnetohydrodynamic)	10,000–30,000	$<10^{-5}$ (present)	$<$0.1

* The values for the arc-heat, and for the two subsequent electrical systems, should be regarded as area figures only.
† Restricted to space propulsion; therefore, no difference between surface and space or low and high expansion ratio of the exhaust nozzle.

The preceding discussion shows that, for take-off from the surface of celestial bodies, the development scope is quite limited, at least in the near future. Even for take-off from the Moon we need an initial acceleration of at least 0.2 $g_\oplus$ which can be produced only by chemical and nuclear-heating systems. We can increase the specific impulse of chemical rockets (at least for upper stages) to some 400 gm sec/gm. We can make the transition to nuclear-heated propulsion systems; but inside the Earth's atmosphere (*i.e.*, below 100,000 feet or at the surface) nuclear propulsion systems offer more difficulties than advantages. Therefore, the use of nuclear propulsion in upper stages of planet-launched vehicles and, of course, its unrestricted use for take-off and landing on the Moon, appears most attractive. *Consequently one can say that, generally, for take-off from and landing on celestial bodies we see, at present, no practical way of breaking out of the* $300<I_{sp}<1,200$ *gm sec/gm bracket.*

The prospects for interorbital propulsion look much brighter, at least in principle. Since much lower accelerations are permissible (and often even desirable) for leaving one orbit in space and transferring to another, why are we concerned about the magnitude of acceleration? The answer is that the concern applies primarily to manned vehicles. If the acceleration becomes too low, the transfer times between Earth and Moon or between

Earth and the planets simply become too long for man and machine to be attractive or even practical. A flight around the Moon and back can, with a chemical rocket, be completed in less than a week. At $n_0 = 5 \cdot 10^{-4}$ g and $I_{sp} = 10{,}000$ sec, the same round trip requires about three weeks. It is doubtful that this can be considered attractive for personnel transport, in spite of the greatly reduced mass ratio. This does not mean that one must blast off to the Moon from an Earth satellite orbit at accelerations comparable to those used at surface take-off (1.3–1.5 $g_\oplus$). In fact, the orbital departure acceleration for Moon-bound spacecraft can be reduced as far as 10^{-2} g without significantly increasing the transfer time or round-trip (mission) period. For this reason, the lower limit of initial acceleration for fast transfers to the Moon was given as 10^{-2} g in Fig. 3.[3] The greater the transfer distance, the lower becomes the initial acceleration still compatible with the desire for fairly fast transfer periods or, even more important, brief mission periods at least for manned flights. The lower limits for the inner solar system (*i.e.*, primarily for reaching the planets Venus and Mars) can be shown to be of the order of 10^{-4} $g_\oplus$, and for reaching the planets of the outer solar system, about 10^{-5} $g_\oplus$.[3,4] These numbers are based on the assumption that the vehicle departs from a satellite orbit close to Earth (*e.g.*, 400–500 miles altitude). In the light of these limitations, the very high specific impulse electrical systems really do not qualify for manned flight missions into the inner solar system, leaving, in principle, only the chemical, nuclear-heating and solar-heating systems. Thus, we are still in the same region of specific impulses as for take-off and landing systems. This is, of course, not the end of the story; but before continuing, let us take a look at the significance of increased specific impulse as to the percentage of expendable material required, since this will furnish us a guide as to the relative importance of high specific impulse for various missions. A measure of the percentage of expendable material required, is the ratio of filled to empty rocket (mass ratio, μ), since $\mu - 1/\mu$ yields the percentage of the over-all initial weight, which is expendable material. For a given specific impulse, the mass ratio is a unique function of the sum-total of all velocity changes of the particular mission, including losses due to gravitational pull or drag during powered maneuvers (ideal mission velocity). Fig. 4 depicts the relation between the over-all (mission) mass ratio and ideal mission velocity for three ranges of specific impulse, corresponding to chemical systems (300–400) nuclear- and solar-heated systems (600–1,000) and electric (*i.e.*, ion and plasma) systems in the range of $6{,}000 \leq I_{sp} \leq 15{,}000$. Superimposed are a number of ideal velocity requirements for lunar and Venus or Mars missions. The latter ones were not computed for minimum-energy transfer with its ensuing mission periods of 2 and 2.66 years, respectively, but for mission periods of the order of one year (including a brief period (30–60 days) of satelliting about the target planet), because such periods appear more practical from the viewpoint of human endurance and equipment reliability. The resulting energy (ideal velocity) requirements are, of course, significantly higher than the minimum values of 34,000 feet per second for Mars and 42,000 feet per second for Venus (from and to Earth satellite orbit). Fig. 4 shows the increasing significance of specific impulse ranges of 800–1,000 at least up to mission class (6). For example for 70,000 feet per second, an $O_2 + H_2$ rocket ($I_{sp} = 400$ gm sec/gm) would require a mass ratio of 250, a nuclear rocket with $I_{sp} = 800$ or 1,000 gm sec/gm would permit a mass ratio of 15 or 8, respectively. If the specific impulse is increased by a factor of 15 over 400, to 6,000, the mass ratio is down to about 1.5. It is worthwhile to note that alone by about doubling the specific impulse attainable with chemical systems one gets the mass ratio down most of the way and, what is most important, from a rather impractical figure to a value which is very well within practical design capability. To reduce the mass ratio further is, of course, *desirable*, but not *vital* for the feasibility of such missions.

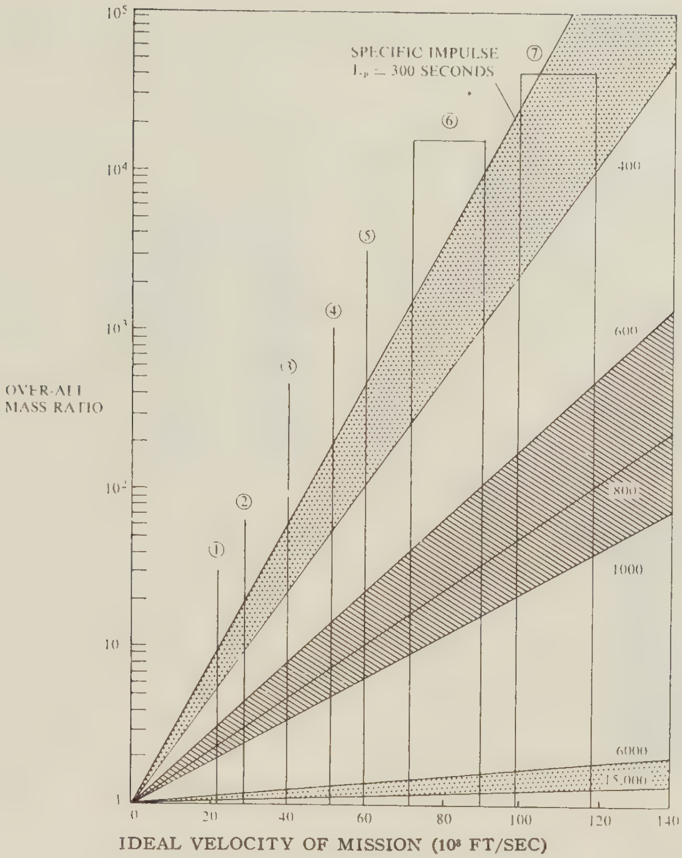
At this point, therefore, we can introduce the distinction between necessary developments to *attain the capability of manned lunar and inner solar system missions* and the developments required for *desirable refinements* in these capabilities and for *attaining the additional capability* of either *transporting very large payload weights* (such as needed for planetary bases) *within the inner solar system*, or of *gaining access to the planets of the outer solar system*. For the first group of achievements, only the development of high-energy chemical propulsion systems (Moon flights) and of nuclear-heating systems (lunar base, fast manned inner solar system reconnaissance flights) is the pace-setter, as far as propulsion is concerned. On the other hand, the second group of achievements depends to a high degree upon the development of electrical propulsion systems, even at the low accelerations indicated above.

This conclusion, presented over a year ago,[3] has been made even more valid by the discovery of the terrestrial radiation belt[5] and the plausible speculation that other planets (if not the Moon), and especially Venus, may be surrounded by similar, deadly radiation belts. These belts make heavy shielding mandatory, to spiral gradu-

[3] K. A. Ehricke, "Comparison of Advanced Propulsion Systems: Solar Heating, Arc Thermodynamics and Arc Magnetohydrodynamics," Convair Rept. AZK-002; December 1, 1957. Presented at the Advanced Propulsion Systems Symposium Rocketdyne, Canoga Park; December, 1957.

[4] W. E. Moeckel, "Nonchemical Propulsion Systems for High-Payload Space Missions," presented at the National Midwestern Meeting of IAS, St. Louis, Mo.; May, 1958.

[5] J. A. Van Allen, *et al.*, "Observation of High Intensity Radiation by Satellites 1958 Alpha and Gamma," IGY Satellite Rept. Ser. No. 3, Natl. Acad. Sci., Washington, D.C.; 1958.



① Moon circumnavigation from and to Earth satellite orbit

② Ascent into Earth satellite orbit

③ Earth satellite orbit to Moon's surface and back

④ Moon circumnavigation from Earth's surface

⑤ Moon landing from Earth's surface and return into Earth satellite orbit

⑥ Round-trip missions to Venus or Mars (no landing), lasting one year or less, start from and return to Earth satellite orbit

⑦ Round-trip missions to Venus or Mars (no landing), lasting one year or less, start from Earth's surface and return to Earth satellite orbit

Fig. 4—Mass ratios for astronautic missions.

ally away from the Earth or close to another planet, as electrical systems would require. In fact, if at least parabolic velocity is to be attained before the vehicle enters the denser radiation belt (500–600 miles altitude), the departure acceleration from low orbits is definitely limited to values not lower than 0.4–0.5 $g_{\oplus}$. So far as we can see today, only chemical and nuclear systems are capable of such accelerations. Slow-spiraling space ships must have heavily shielded life support systems, capable of protecting the crew for weeks or months in terrestrial or other planetary radiation belts. This lowers the useful payload capability of those ships. An alternate possibility is orbital assembly far out in cislunar space at 50,000–60,000 miles distance.[6] Such assembly orbits for large electrical space ships are very expensive for the lower-impulse high-acceleration systems. The energy required to climb into a circular orbit at 150,000 miles altitude is equivalent to the energy required to enter a very fast transfer orbit to Mars (130–140 days transfer time instead of 250 days for minimum energy). To make such supply operations from Earth practical requires nuclear propulsion; and then, of course, the question arises why one should not keep on going and carry the entire mission out with the nuclear-heating system. This system, most likely, might have to be carried along anyway in a number of missions for fast maneuvers in the vicinity of the target planet and its radiation belt.

Therefore, if radiation belts are indeed limited to the proximity of planets, the discovery of the terrestrial radiation belt has emphasized the importance of relatively high accelerations over very high specific im-

[6] The radiation belt seems to fade away at lower altitudes, but the assembly orbit would have to be higher, in order to assure a high-speed transfer ellipse through the radiation belt. Still higher orbits would be subject to undesirably strong lunar perturbation.

pulses. On this premise it is of utmost importance for the advancement of the U. S. manned space flight capability to concentrate on the development of a nuclear-heating space propulsion system.

This conclusion does not diminish the importance of very-high-I_{sp} electric propulsion systems in general. On the above premise regarding the distribution of radiation belts, however, their importance lies not in their application as main propulsion systems for manned vehicles of the next 10–15 years, but in their significance for what comes thereafter. If, on the other hand, radiation belts of intensity dangerous to man should be discovered in interplanetary space, then it will become necessary from the very beginning to rely on very-high-I_{sp} electric propulsion systems, because short-time passage through such presumably quite extended heliocentric belts would require parabolic or hyperbolic velocities with respect to the Sun. This would tax the capability of the nuclear-heated system so severely that payloads again would become very small, or the use of this system for interplanetary missions even become impractical altogether. In this case, the very-high-I_{sp} system would remain as the only hope for ever achieving interplanetary flight (even if its acceleration level would remain very low), because its enormous payload capability allows for adequate weight of shielding material to protect the crew. At present, we have no proof of the existence of such radiation belts in interplanetary space. Available theoretical and recorded evidence (Pioneer I, III, IV) suggests that interplanetary space (at least down to a yet unspecified distance from the Sun) is "clean." However, the need for obtaining conclusive proof soon is very urgent, in order to enable the adoption of the correct development philosophy in the area of space propulsion systems and space vehicles in general. This underlines the great importance of systematic exploration, as soon as possible, of the entire inner solar system with interplanetary and planetary probes.

Thus, we must eventually find a way of developing the best of the various very-high-I_{sp} electrical propulsion systems to a level of perfection where they become of real, practical importance to astronautics. At present, the work is one of research, concerned with energy conversion, solid-state physics, electronics and nuclear research. However, by regarding it as no less important than our present front line in space technological development, we plant the seeds for man's eventual freedom of operation in the entire solar system and beyond.

Contributors

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While with the Manhattan Project in 1944, he assisted in the operation and maintenance of a cyclotron and investigated and identified several radioactive isotopes. At the Los Alamos Laboratory, N. M., in 1945, he measured the fast neutron energy distribution in fast reactors.

He became an assistant professor at the University of Rochester, N. Y., in 1951, and spent a major part of his time on investigations of pi mesonic atoms and pi meson scattering from hydrogen.

He joined Avco Research Laboratory, Everett, Mass., in 1956, where he is doing research in X-ray and ultraviolet radiations in the field of magnetohydrodynamics.

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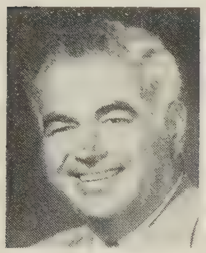
Arthur Kantrowitz was born on October 20, 1913, in New York, N. Y. He received the Ph.D. degree in physics in 1947 from Columbia University, New York, N. Y. He is a recipient of the Fulbright Scholarship to Cambridge and Manchester, and a Guggenheim fellowship.



A. Kantrowitz

His experience includes many years as a professor at the Cornell University Graduate School of Aeronautical Engineering and as head of the Gas Dynamics Section of the National Advisory Committee for Aeronautics.

Under his guidance the Avco Research Laboratory, Everett, Mass., has been experimenting in this field for over two years and has developed a broad knowledge in the fundamentals of the area. He is vice-president and member of the Board of Directors of Avco.

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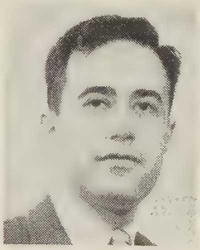
Harry E. Petschek was born on September 12, 1930 in Prague, Czechoslovakia. He received both the B.S. and Ph.D. degrees in engineering physics from Cornell University, Ithaca, N. Y. While at Cornell he was principally concerned with the study of high temperature gases in shock tubes.



H. E. Petschek

After receiving the Ph.D. degree he was appointed to the Physics Department at Princeton University, N. J., where he conducted theoretical research in magnetohydrodynamics.

He joined Avco Research Laboratory, Everett, Mass., in 1956, and has been performing significant work in the field of high temperature gas dynamics and magnetohydrodynamics.

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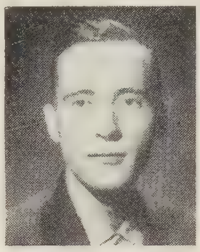
Warren D. Rayle was born on November 30, 1925, in Leipsic, Ohio. In 1942 he entered Findlay College, Ohio. Subsequently he attended Baldwin-Wallace College, Berea, Ohio, and Case Institute of Technology, Cleveland, Ohio, under the auspices of the Navy V-5 and V-12 programs. After serving for about seven months, he was released from active duty. Returning to Case, he received the B.S. degree in chemical engineering in January, 1948.



W. D. Rayle

He then joined the NACA Lewis Laboratory in Cleveland, where his work included research in the relationship between fuel properties and efficiency and ignition characteristics of turbojet combustors, study of the mechanism of screeching combustion in jet engines, and the design and testing of various experimental combustors for turbo- and ram-jet applications. Early in 1957 he was appointed head of the high energy fuels section.

With the increased interest in space flight possibilities, emphasis began to shift to possible propulsion systems for space, and research was initiated into various electrical propulsion possibilities.

When the NASA was initiated, Mr. Rayle became head of the plasma propulsion section in the newly organized Advanced Propulsion Division.

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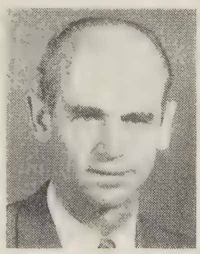
Ernst Stuhlinger (A'54) was born in Niederrimbach, Germany, December, 1913. He attended schools at Tuebingen and received the Ph.D. degree in physics at the University of Tuebingen in 1936.



E. Stuhlinger

He was appointed assistant professor in the physics department of the Berlin Institute of Technology and was a member of the faculty from 1936 to 1941. He worked closely with Dr. Hans Geiger, developer of the Geiger counter, for seven years. From 1939 to 1941, he was a

member of a research group conducting studies in nuclear physics. In the spring of 1943 he joined the Rocket Development Center at Peenemuende, which was under the technical supervision of Dr. Wernher von Braun, presently Director of Development Operations Division, Army Ballistic Missile Agency. There he carried on research in connection with the development of guidance and control systems for the V-2 guided missile.

He came to the United States in 1946 under the auspices of the U. S. Army Ordnance Corps. He conducted research and development work with guided missiles at Ft. Bliss, Tex., and assisted in high altitude research firings of captured V-2's at White Sands Proving Ground, New Mexico. Since 1950 he has been a member of the guided missile team at Redstone Arsenal, Huntsville, Ala.

At present he is Director of the Research Projects Laboratory.

Dr. Stuhlinger has gained recognition in recent years for his feasibility and design studies of electrical propulsion systems for space ships.

R. N. SEITZ

Robert N. Seitz was born in Willoughby, Ohio, on July 21, 1929. He received the B.S. degree in physics in 1954 from Ohio State University, Columbus.

After some experience in electron emission work, he joined the staff of Project Doanbrook at Case Institute of Technology, Cleveland, Ohio, in 1956. There he was primarily concerned with abstract-network studies. He received the M.S. degree in physics from Case in 1958, and is presently employed by the Army Ballistic Missile Agency, where he is engaged in ionic propulsion studies.

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Albert E. von Doenhoff was born on June 16, 1910, in New York, N. Y. He received the B.S. degree in electrical engineering in 1930 and the E.E. degree in 1931 from Columbia University, New York, N. Y.

He is presently at NASA Headquarters, Washington, D. C., as a specialist on advanced propulsion systems. He had been associated with the Langley Research Center of the NASA, in Virginia, for many years as an aeronautical research scientist. His work was important in deriving the NACA airfoils, and made substantial contributions to the knowledge of boundary layers. Recently, he has turned his attention to the use of solar energy for space vehicles. He is in the midst of the problem to determine the most suitable form of solar energy conversion device, both for the present and for the immediately foreseeable future.

He is a member of the Institute of Aeronautical Sciences, Sigma Xi, and the Scientific Research Society of America.

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